JOURNAL OF THE AMERICAN ROCKET SOCIETY

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Rockets as Research Tools in Aeronautics, by H. L. Dryden 3
Metal Parts for Solid Propellant Rockets, by L. G. Bonner 9
Some Observations on the Problems of Rocket-Motor Cooling Design, by J. L. B. Selwood 16
The Prospects of Jet-Reaction Flight, by Eugen Saenger 27
U. S. Rocket Development 42
ARS Elects First Fellow Member 43
Jet Propulsion Centers and Fellowships Announced ... 44
ARS News 47

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NUMBER 76 **MARCH, 1949**

ROBERTSON YOUNGQUIST, Editor

The JOURNAL OF THE AMERICAN ROCKET SOCIETY is devoted to disseminating information on the development of rocket and jet propulsion by printing original technical papers on jet propulsion, data on the latest experimental developments, historical notes, patent specifications, reviews of books and current literature, and news of the Society and individual members.

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INYOKERN, CAL.—FEBRUARY 1, 1948

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JOURNAL OF THE
AMERICAN ROCKET SOCIETY

Number 76 ROBERTSON YOUNGQUIST, *Editor* March, 1949

ROCKETS AS RESEARCH TOOLS IN AERONAUTICS

By Hugh L. Dryden

Mem. ASME, Director of Aeronautical Research, National Advisory Committee for Aeronautics, Washington, D. C.

ONE of the important developments of World War II was the application of rockets as military weapons which led to an active development program on rocket propulsion. The most impressive rocket weapon was, of course, the 12-ton V-2 rocket, the result of scientific and engineering activities dating back to 1934. The place of rocket propulsion in projectiles and missiles is now well established.

Since the war other applications of rockets have been made and it is three of these that I wish to treat briefly. All three involve the use of rockets as research tools in aeronautics. Members of The American Society of Mechanical Engineers Aviation Division and of the American Rocket Society will perhaps be interested in this nontechnical account of rockets at work today in the advancement of aeronautical science. The three applications are to the propulsion of models in free flight for aerodynamic measurements at high speeds, to the propulsion of piloted research aircraft, and to the sounding of the upper atmosphere.

The development of jet propulsion completely revolutionized the art of aircraft design and made possible a great advance in the performance of aircraft especially as regards speed. In particular, aircraft speeds almost immediately entered the so-called transonic or mixed-flow region in which the air flow begins to change from the subsonic to the supersonic type. For years it had been supposed that aircraft could not penetrate beyond a "sonic barrier," the region where the drag of the aircraft increased disproportionately. Early experience with the severe trim changes and large stick forces encountered as the speed was increased led to the picture of the transonic region as a region to be avoided or hurriedly traversed. The flight of the Bell X-1 to supersonic speeds has modified this concept. In particular, a look at the quantitative information obtained led to the exercise of engineering ingenuity to find methods of dealing with or minimizing the difficulties encountered. As is always the case, knowledge dissipated fear. The pendulum has now swung. Designers are clamor-

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ing for specific information on aerodynamic and propulsion characteristics of all possible configurations at transonic speeds so that practical aircraft can be built to fly in and through this region.

NACA Anticipates Needs

The National Advisory Committee for Aeronautics and other research agencies anticipated this need by a few years and undertook to develop techniques for research in this field. Unfortunately wind tunnels, the most useful research tools at subsonic speeds, have limitations at speeds at and near the speed of sound because the passage chokes, blocking further speed increase when the speed at the narrowest cross section reaches the speed of sound. There is a blind spot whose width is dependent on the size of model and on its shape.

One of the early techniques was to drop heavily weighted models from an aircraft at high altitude. If the weight per unit frontal area is sufficiently great, speeds up to and slightly exceeding the speed of sound can be obtained. By telemetering accelerometer readings and by radar tracking the drag can be measured.

Another of the earliest techniques, the wing-flow method, was developed by R. Gilruth and his co-workers at the NACA Langley Aeronautical Laboratory. Gilruth took note of the increase in local air speed above the curved upper surface of an airplane wing in flight. The local speed may be one and one-half or more times the flight speed so that there is a region near the wing of an airplane flying at 0.7 the speed of sound where the local speed exceeds the speed of sound. While there are speed gradients present, the speed is approximately uniform over a region of sufficient size to include a small-scale model. Gilruth mounted small models on a balance built into the upper surface of a fighter airplane which could be dived at high speed. While the models are small and the conditions not ideal, for a time this and the dropped-body technique were the only methods of study in the transonic region. The method yielded information of great value and is still useful.

Shortly after the wing-flow method was in use, the same general principle was applied to wind tunnels by the Lockheed Company in the GALCIT Co-Operative Wind Tunnel and by the NACA in the Langley 7 by 10-Foot Wind Tunnel. The local transonic region was created by installing a "bump" on the wall of the wind tunnel; hence the name "bump method."

Rocket-Propelled Models Introduced

These methods have now been supplemented by another method, the rocket-propelled model, made possible by the general availability of rockets. In 1945 the NACA established a field test station for this purpose

at Wallops Island, Va., on the Atlantic Coast south of the Naval Ordnance Test Station at Chincoteague. The information obtained by this method has been found so useful to aircraft designers that the station has continued to grow and a committee of aircraft designers recommended recently that its activities be increased threefold.

The research models are propelled by standard rocket motors rebuilt to specifications of size and performance suitable for the particular aerodynamic models they propel. Much of the work has been done with standard $3^1/_2$ and 5-in. solid-propellant rockets used by the military services to propel explosive charges. The rockets are modified by changes in the nozzle and igniter to give the desired thrust and operating time and by adding tailpipes as needed. Often two rockets are used in sequence as booster and sustainer.

The simplest type of research model is that used for measurements of drag at zero lift. This model consists of a standard rocket placed inside a light wooden body or fuselage which may or may not carry wings, fins, or wings and tail of the configurations to be studied. The model itself contains no instrumentation. A typical drag model is about 5 in. in diam and 56 in. long and weighs 35 to 45 lb at firing. With a second booster rocket a speed of about 1.9 times the speed of sound is reached; without it about 1.2 to 1.5 times the speed of sound, depending on the weight. The distance from the ground station to the rocket is recorded by means of a Doppler radar and from the record the speed and acceleration or deceleration can be accurately determined. The drag is determined from the deceleration of the missile after the rocket has burned out. The effects of wing planform, thickness, and aspect ratio have been determined for many configurations, including sweptback and triangular wings.

Measurement of Control-Surface Effectiveness

A less simple model is used to measure control-surface effectiveness. This model carries a small radio transmitter which transmits a continuous signal from air antenna which has marked directional characteristics, this instrumentation occupying the nose. These models are somewhat heavier than the drag models, weighing 100 to 120 lb. The wings are equipped with trailing-edge flaps which are deflected in opposite directions on the two sides, thus causing the model to spin. Because of the directional characteristics of the antenna the signal received on the ground fluctuates in intensity, the number of fluctuations per sec corresponding to the rate of rotation, which is a measure of control effectiveness. In some instances the spin is observed to slow down and change sign as the speed is increased or decreased through a certain range. Such a result indicates a reversal of control effectiveness in certain speed ranges for that type of control-surface design.

Other missiles carry more complex instrumentation and permit other types of measurement. Telemetering is used to transmit the measurements to the ground. Longitudinal acceleration, lateral acceleration, impact pressure, control-surface position, and hinge moments of control surfaces are some of the quantities which can be measured. Programing devices may be used to oscillate the control surfaces or to displace them suddenly and repeatedly. By such methods information can be obtained about longitudinal and lateral stability, damping and period of oscillations, effectiveness of controls, pressure distribution, and similar aerodynamic characteristics as a function of speed through the transonic region. The models may represent the configurations of piloted aircraft or missiles intended to be constructed.

Rocket-propelled models may be used for many other special investigations, for example, the study of wing flutter and the study of aerodynamic heating. An important problem under study by this method is that of pilot escape from transonic and supersonic aircraft. The nose section containing the cockpit may be released at high speed and its motion and acceleration telemetered to the ground. The techniques are constantly being developed to permit the study of many of the new problems of high-speed flight.

Rocket Motors for Aircraft

The second application of rockets is to the propulsion of piloted aircraft, specifically to the research airplanes. The concept of building a special research airplane for the sole purpose of making quantitative measurements at very high speeds arose, probably in many minds, in the year 1944. The proposal was vigorously advocated by John Stack and discussed within the NACA staff and the NACA Aerodynamics Committee. Within a year the military services were engaged in receiving proposals and a coordinated program was arranged, involving the study of conventional and sweptback wings of two thicknesses and the use of turbojet and rocket power. The Navy sponsored the Douglas D-558 series, the Air Force the Bell X-1 series. The D-558 early captured the speed record, since exceeded by the F-86, and the X-1 was flown faster than sound. Striking as these performances are, these accomplishments were but incidental to the chief purpose and are not nearly so significant as the quantitative data already obtained and that now being obtained on drag, stability, trim changes, air loads, etc., at all speeds within the capabilities of the aircraft.

Misconceptions About the X-1

There are many misconceptions about the X-1. It is not a tactically useful airplane; it was designed for 500 lb of instruments, strength to withstand a load of 18 times its weight; it was kept small so that control

forces could be kept within the capabilities of the pilot; and it can fly at full power for only a few minutes. Its aerodynamic design was not regarded as optimum for supersonic speeds; it was built to verify results obtained by other methods and to study further the behavior of conventional configurations. Only a rocket could furnish sufficient thrust to drive this configuration at supersonic speeds and then only at very high altitudes where the drag is reduced because of the reduced air density.

It seems unlikely that rocket engines will be used as the main power plant of useful piloted aircraft because of the tremendous fuel consumption though they will probably be used as auxiliary power plants for take-off and combat.

The following information about the rocket has been published. The rocket engine is Reaction Motors RMI-6000C4, having a total thrust of 6000 lb with engine weight of 210 lb. The X-1 carries 8177 lb of fuel (61 per cent of the gross weight of the airplane) which at full power is consumed in 2.5 minutes (54 lb/sec). The engine uses alcohol and liquid oxygen which are forced through the supply nozzles by the pressure from compressed nitrogen gas.

The engine consists of four separate chambers grouped into a single unit. By this means the pilot may select a thrust of 1500, 3000, 4500, or 6000 lb at will. He may also turn any tube on or off at will and thus has a 4-step control until the fuel is exhausted. The chamber pressure is 230 psi and the combustion temperature is about 5000 R. To cool the chamber the alcohol fuel is mixed with about 25 per cent water and circulated through a cooling jacket around the nozzle before passing to the combustion chamber. Addition of the water is said to have only a slight effect on the thrust.

The first flight of the XS-1 with this rocket engine was made on Dec. 9, 1946, a Mach number of 0.79 being reached at 35,000 ft with half power. Since then the airplane has made many flights in the hands of several pilots.

Rockets for Auxiliary Power

Other research airplanes will use rocket engines as auxiliary power plants to enable the attainment of still higher speeds. I should like to emphasize again that the interest is not in speed records but in research data throughout the speed range. The engineer is concerned with that region within which large changes in characteristics occur, which is different for different airplanes in the series. The expected phenomena are known to some extent from wind-tunnel data, dropped-body tests, wing-flow tests, and rocket-propelled-model measurements. Research airplanes give the final check and give confidence to designers to proceed with practically useful aircraft capable of flight in the transonic and supersonic regions.

Sounding Rockets

A third application of rockets as research tools in aeronautics is to the study of conditions in the upper atmosphere. Interest in the upper atmosphere arises from many sources, for the conditions there have a profound influence on human affairs. Astronomers have been annoyed with the interference of the atmosphere in their observations, for even in the absence of clouds an ozone curtain shuts off a view of the short-wavelength end of the spectrum of stars and sun and so denies them some information on the constitution and physical state of the heavenly bodies.

Ordinary men are thankful for the existence of the curtain because it protects them from the harmful radiations of the sun. Communication engineers know the interruption to the transmission of speech and code messages when the upper atmosphere is disturbed. The commercial value of uninterrupted communication added to the curiosity of scientists and led to intensive study of the upper atmosphere by many indirect means.

Rockets in the course of their military uses actually propelled vehicles to high altitudes and the knowledge and experience gained in their development makes possible the design of special sounding rockets for the direct study of physical phenomena at high altitudes. A new incentive for such study has also arisen. The air-breathing forms of jet propulsion consume fuel at a much smaller rate than the rocket, since the oxygen then comes from the atmospheric air. Air-breathing engines should therefore be used wherever possible. Economic travel at supersonic speeds will be possible only at high altitudes. However, the difficulty of keeping the fire burning in an air-breathing jet-propulsion engine increases as the altitude of operation is increased. To study these problems, as well as the lift, drag, and stability of vehicles, we need to know the physical conditions so that they can be reproduced on the ground in laboratory-research equipment.

One of the principal uses of the captured V-2 rockets which are being fired by the Army Ordnance Department at White Sands is this type of study. A great many agencies have co-operated in the work. Special sounding rockets have now been developed and the work will proceed at a greater rate. The NACA is engaged in the problem of reproducing the physical conditions of interest for studies of aerodynamic and combustion phenomena. One of its many advisory subcommittees serves as a meeting place for exchange of scientific data on problems of the upper atmosphere.

I will leave to others the prophecy of the future uses of rockets. What I have told you is what rockets are now doing to advance aeronautics.

METAL PARTS FOR SOLID PROPELLANT ROCKETS

By L. G. Bonner

Technical Director, Allegheny Ballistics Laboratory, Hercules Powder Company, Cumberland, Md.

THE field of design and fabrication of metal parts for solid-propellant rockets poses a set of problems which are probably unique in the world of mechanical engineering. This uniqueness arises primarily from two sources—the frequently extreme emphasis on minimum weight for all components and the fact that the total useful life of the assembly is by ordinary standards exceedingly short. To date this set of problems has been solved to only a small degree and it is the purpose of this paper to present primarily a survey of what yet remains to be done rather than a review of what has already been accomplished.

The simplest description of a solid-fuel rocket would be as a high-pressure boiler with a hole in one end. I believe almost anyone would admit, though, that it is probably not desirable to make this boiler according to the ASME Code. Obviously the weight of the assembly is important. The big question is: How important? The answer to this depends more or less on the application. For purposes such as assisting the take-off of aircraft, weight is of secondary importance since, in any event, the total rocket weight is negligible in comparison with the weight of the load it is moving. For the more interesting case of a free-flight rocket, I believe it is clear that for a given propellant charge and specified final velocity, each pound that can be knocked off the rocket chamber may be replaced by a pound of useful pay load. The other virtues of improved rocket efficiency are less obvious

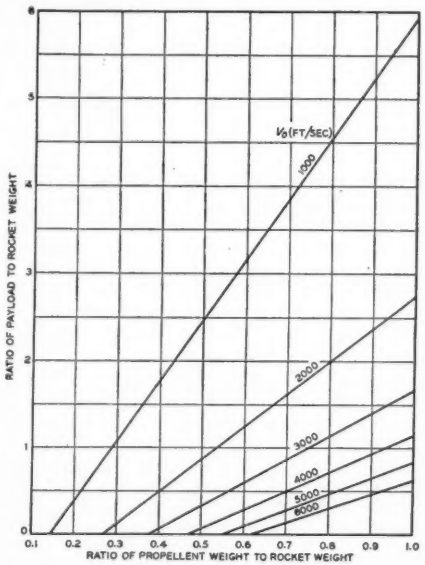


FIG. 1 RATIO OF PAY LOAD TO ROCKET WEIGHT PLOTTED AGAINST RATIO OF PROPELLANT WEIGHT TO ROCKET WEIGHT

Presented at the 1948 Annual Meeting of the American Rocket Society, Hotel Statler, New York, N. Y., Dec. 2, 1948.

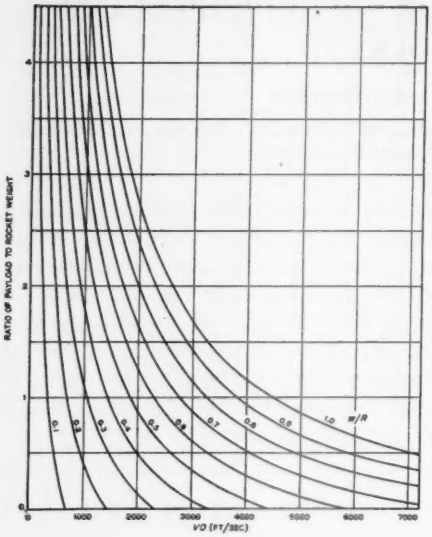


FIG. 2 RATIO OF PAY LOAD TO ROCKET WEIGHT PLOTTED AGAINST FINAL VELOCITY

but I have made an effort to present at least some aspects in Figs. 1 and 2. In Fig. 1 I have plotted the ratio of pay load to rocket weight against the ratio of propellant weight to rocket weight for various values of V_o, the final velocity in the absence of aerodynamic drag. In Fig. 2, the ratio of pay load to rocket weight is plotted against V_o for various values of the propellant- to rocket-weight ratio. In order to examine this effect a little more closely, consider the effect of reducing by 10 per cent the weight of the inert components of a rocket originally constructed with a propellant- to rocket-weight ratio of 0.5. The effect on pay load at constant rocket weight or on rocket weight at constant pay load for various final velocities is shown in Table 1.

TABLE 1 EFFECT OF 10 PER CENT REDUCTION IN METAL PARTS WEIGHT ON PAY LOAD AND ROCKET WEIGHT

V_o	1000 fps	2000 fps	4000 fps
Increase in M_p at constant R.....	7.3 per cent	11.4 per cent	73 per cent
Reduction in R at constant M_p.....	6.8 per cent	10.2 per cent	42 per cent

On the other hand, if it is elected to keep the pay load and propellant weight constant, the effect on final velocity for various values of pay load to rocket-weight ratio is shown in Table 2.

TABLE 2 EFFECT OF 10 PER CENT REDUCTION IN METAL PARTS WEIGHT ON FINAL VELOCITY

Initial M_p/R	Initial V_o	Final V_o
0.1	3880 fps	4140 fps
0.5	2600 fps	2710 fps
1.0	1840 fps	1895 fps
2.0	1170 fps	1190 fps

I believe these figures make it abundantly clear that the virtue of reduced weight of rocket components depends quite strongly on the nature of the application of the rocket. For a low velocity and high pay load, unit weight is relatively unimportant but for a high-velocity high-performance round, it becomes very critical indeed and, in fact, may make all the difference between success and failure. For the high-performance rocket, then, the principal problem becomes that of how to attain the necessary minimum weight of inert components. We will attack this problem under the two headings: Materials and Design. These are not completely separable as we shall see later but this will do for a starting point.

Materials

Under materials, the first point to be considered is that of a suitable set of criteria for relative evaluation. One criterion obviously is density, a low-density material being inherently more desirable for light-weight applications. The other criterion is strength and this is subject to considerable latitude in interpretation. I should like to propose that the yield point of the material as determined by standard test practices be at least tentatively accepted as an index of comparison for a very practical reason. It is certainly true now, and will probably continue to be true for quite some time, that every completed rocket-chamber assembly of highly refined design must be subjected to a hydrostatic test before use. Since respectable ballistic behavior requires that there be no relative distortion of parts before or during use, this hydrostatic test cannot be carried beyond the yield point of the material. If we accept this criterion, the design condition becomes that at which the yield stress is equaled by the stress in the material at the maximum expected rocket-chamber pressure in normal operation multiplied by a factor of safety as dictated by considerations of predictability of propellant performance and of exposure of personnel and equipment. Thus, for a well-established propellant or propellant charge in a rocket for an application involving negligible personnel exposure, the stress at the maximum expected pressure might conceivably be set equal to the yield stress of the material. For a less well-known propellant or for an application such as JATO in which exposure is high, this stress might be reduced to $1/2$ to $1/3$ of yield. In any case the hydrostatic test would be performed at the yield stress in order to assure realization of the full safety factor. All of the foregoing discussion is applicable really to the rocket case only. Material criteria for the nozzle are inherently different in nature and will be taken up in due course.

With all of this in mind, we might proceed to list possible classes of materials as follows: Carbon steels; alloy steels, normalized; alloy steels, heat-treated; stainless steels; light metal alloys; titanium; and nonmetals.

The carbon steels, I believe, we may dismiss at once. Alloy steels in the normalized condition are probably currently the most available and the most useful materials for rocket chambers in sizes above the seamless tube range. Of course, in this range use of at least moderately heat-treated materials would be desirable, but a problem not yet completely solved is that of avoiding distortion during heat-treating, especially at the small wall-to-diameter ratios characteristic of high-performance rockets. Even in the seamless-tube range heat-treating cannot be used as extensively as might be desired; again, for reasons of wall thickness. For example, a 12-in. diam tube with a design pressure of 2000 psi would have the following wall thickness to yield point relations:

Yield Point	Wall
40,000 psi	0.30 in.
60,000 psi	0.20 in.
80,000 psi	0.15 in.
120,000 psi	0.10 in.

The thinner of these walls are, to the best of my knowledge, not producible in seamless tubes even at relatively small diameter and, even if available, might not be particularly useful because of the problem of threading or otherwise attaching end fittings.

The stainless steels as a class are not particularly useful because of their extremely low values of yield. There are a few exceptions to this as some of the stainless steels are outstanding in their ability to retain strength at very high temperature and their use is certainly indicated in any case in which this situation is encountered. The stainless steels, too, provide an excellent example of another phenomenon, the dependence of physical properties on rate of loading. In Fig. 3 the stress versus strain relations

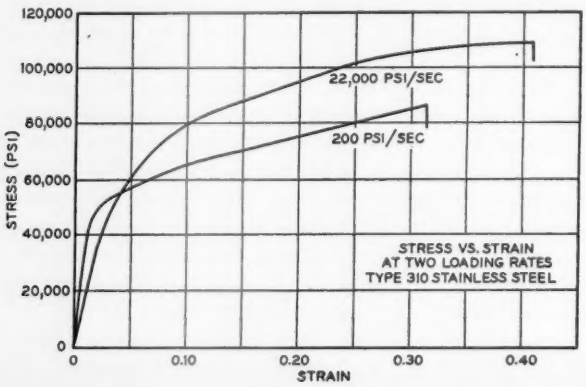


FIG. 3 STRESS-STRAIN RELATIONS FOR USS TYPE 310 STAINLESS STEEL FOR TWO STRESS RATES

for USS Type 310 stainless steel are plotted for two stress rates, one that of the ordinary testing machine and the other approximately 100 times greater. The exact shape of the lower portion of the high-rate curve is probably not significant owing to instrument difficulties, but the increase in yield point and in ultimate strength almost certainly is both real and significant. This would seem to indicate that, for rockets whose duration is a few seconds or less, the design criterion could be based on a high-rate yield value at a substantial saving in weight, provided that a "hydrodynamic" proof test were devised to replace the present hydrostatic test. Unfortunately the possible saving is much less with more nearly perfectly elastic materials, but the subject is still of enough interest to merit careful examination.

Aluminum Alloys

Many of the aluminum alloys, particularly the 24 and 75ST varieties, are exceptionally attractive for their high strength-to-weight ratio. For example, 75ST may attain a yield of 70,000 psi and has a density approximately $^1/_3$ that of steel. These alloys, however, have serious disadvantages; namely, the impossibility of recovering the physical properties after the welding that would be essential in the fabrication of tubes larger than those that can currently be drawn and the fact that strength begins to drop rapidly with only moderate heating of the material. This means that steps must be taken to insulate effectively the inner aluminum wall from the hot gases resulting from propellant combustion. This poses some difficult and in many cases insuperable propellant-design problems.

Titanium is included in this list only as a sample of what may be available in the future. Production and fabrication of the pure metal is currently being explored by E. I. du Pont de Nemours and Company and by Remington Arms Company and the limited data currently at hand would seem to indicate that, when available in quantity, it may have even more attractive strength-to-weight ratios than aluminum and will suffer less from temperature effects.

Under nonmetals, I refer primarily to the fabric or fiberglass reinforced plastic type of construction. The strength-to-weight possibilities of this class of material have been amply demonstrated but no satisfactory method has yet been proposed for attaching metal end pieces to the plastic tube and I believe it safe to say that the possibility of use of an integral plastic nozzle is quite remote.

This brings us to the subject of nozzles. The principal design criterion here seems to be less that of physical strength than of resistance to the phenomenon known as erosion. Erosion seems to be primarily the result of surface melting due to the high temperature and high velocity of gas flow in the Venturi section and is most effectively combatted either by

use of a metal with the proper combination of heat capacity and conductivity or by selection of a material with a melting point substantially above the temperature of the gas. Successful nozzles of the first class have been made from copper which in spite of its low melting point is able to conduct heat away from the zone of transfer at a sufficiently rapid rate, provided the mass of the piece is sufficiently large. This does not make for an efficient rocket, however, and the attempt to use steel is usually made. In this application the low-carbon steels have shown themselves to be by far the most satisfactory. The alloy steels are considerably less desirable and the stainless steels are even worse. This variation presumedly arises from the fact that any alloying material must, of course, reduce the melting point of the alloy and, in steels at least, it also reduces the heat conductivity. For those cases where, because of high gas temperature or long duration, low-carbon steel is not satisfactory, the second alternative must be resorted to. Both graphite and molybdenum, suitably supported, have proved adequate for such applications and there has even been some limited success with ceramic structures.

Design Considerations

If we pass now from the subject of materials to that of design we find that aside from considerations inherent in the fabrication problems already discussed, there is one principal problem; namely, that of attaching nozzles and end caps to a cylindrical body. In general one or the other may be constructed integrally with or welded to the body but some provision must be made for attaching the other since it seems to be essential to introduce the propellant charge at some stage in the process and the welding of attachments to loaded chambers is not in general recommended. The ideal end-closure attachment system would be one which involved no reduction in ID or increase in OD beyond the values for the tube section itself and, when completed, was light in weight, developed the full strength of the tube material, and involved no critical machine work. This ideal has not yet been attained but Fig. 4 shows what might be considered some

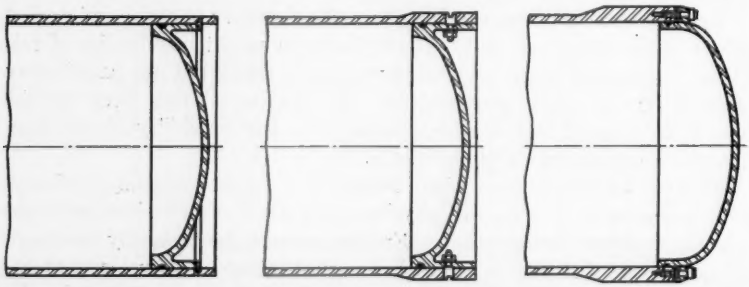


FIG. 4 TYPICAL END-CAP ATTACHMENTS

successive approximations to this state. I have omitted the threaded joint on the grounds of its familiarity and its failure to approach the ideal closely. The third example is that of the bolting-flange arrangement, the second shows a radial pinning system, and the first involves the use of a snap ring. In all cases, the necessary thickening of the wall has been shown as taken up on the outside, but it could just as well be taken on the inside, if propellant loading conditions permit. It will be noted that use is made in all three examples of the "O" ring seal to keep pressure stresses away from critical points and to prevent the leakage of gas.

Finally, I should like to add a word about the economics of rocket construction. An attempt can be made to plot the relative cost of chamber fabrication against what might be called the efficiency index of the rocket. The specific shape is assumed but the general features of the curve are real. Minimum cost for a given propellant charge will be achieved at some intermediate propellant- to rocket-weight ratio. The precise value of this minimum will depend on various factors such as the size of the rocket, the materials used, etc. The left-hand side of the curve is almost completely devoid of interest since it represents primarily the increase in cost due to the sheer weight of material used. The right-hand side, however, reflects the effect of using higher priced materials, more elaborate fabrication processes, and more precise machine work. How far to the right-hand side of the minimum it is economically feasible to go depends in turn on a variety of factors such as cost of the propellant that would be eliminated by the use of smaller and more efficient rockets, handling considerations, and required performance specifications. I am sure there is no unique answer to this question applicable to all rockets but I am equally sure that it is a factor that must be considered in parallel with all the other factors affecting the final design of a rocket for a specific application.

ARS TECHNICAL SESSION AT ASME 1949 SPRING MEETING

THE American Rocket Society will sponsor a technical session at the 1949 Spring Meeting of The American Society of Mechanical Engineers to be held at the Mohican Hotel, New London, Conn., May 2–4, 1949. The ARS session will be held on Monday, May 2, at 8 p.m. The following two papers will be presented:

"Under-Water Jet-Propulsion Systems," by C. A. Gongwer, senior design engineer, Aerojet Engineering Corporation, Azusa, Calif.

"Torpedo Propulsion System," by Frederick Maxfield, technical director, Section RE6A, Bureau of Ordnance, Washington, D. C.

SOME OBSERVATIONS ON THE PROBLEMS OF ROCKET-MOTOR COOLING DESIGN

By J. L. B. Selwood

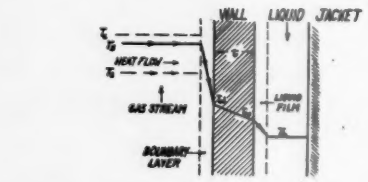
Bell Aircraft Corporation, Buffalo, N. Y.

Introduction

THE high temperature, velocity, and density of the gases in a liquid-propellant rocket-motor combustion chamber require some form of forced convection cooling on most motors of a duration greater than a few seconds. The quantity of heat transfer is too great to permit the economical use of a self-contained circulatory cooling system. One or both of the rocket propellants is used. In this paper the various methods of cooling and the design parameters are outlined, and methods of improving the present technique are discussed. Risk of breakdown in cooling is a major source of unreliability in rocket motors and considerable improvement is possible.

A regeneratively cooled motor consists of a thin-walled combustion chamber and a supersonic nozzle, both surrounded by an outer jacket which forms the cooling passage. One of the propellants is introduced into this annular passage at the nozzle exit and flows along the nozzle and combustion chamber, absorbing heat from the wall. The propellant is then injected into the combustion chamber, and the heat absorbed by it during cooling of the walls is added to the heat of combustion of the propellants. There is, therefore, no loss in performance due to the cooling process.

Regenerative cooling of the rocket wall is possible only because of the existence of a relatively stagnant gas film or boundary layer between the wall and the main gas stream.



(4) $q/a = h_g(T_g - T_l)$ (8) $T_l - T_c = \frac{q}{a}(\frac{t}{k})$

(5) $q/a = \sigma \epsilon_l \epsilon_g (T_g^4 - T_l^4)$ (10) $h_c = \frac{a}{d(T_c - T_l)}$

(6) $q/a = q_{rad} + q_{conv}$ (13) $A = \frac{w}{\rho v}$

(7) $\Delta H = q_{tot}(\frac{dt}{t})$ (14) $\Delta P = \frac{f L \rho v^2}{2 w_H}$

FIG. 1 HEAT FLOW THROUGH ROCKET WALL

Heat Transfer Process

Fig. 1 is a line diagram of the heat-transfer process. The combustion gases are at a high temperature. There must be a large temperature drop through the boundary layer to avoid overheating the metal wall. Heat is transferred by radiation (Equation [5])[1] and convection (Equation [4])[1]

Presented at a meeting of the Metropolitan Section of the American Rocket Society Nov. 19, 1948.

[1] Equations shown in Fig. 1.

in achieving this temperature drop. The problem is to determine its magnitude.

To maintain heat flow through the wall, there must be a temperature gradient. The problem is to determine the wall temperature at the gas and coolant interfaces (Equation [9]).[1]

The heat is finally carried away by the coolant (Equation [11]).[1] When cooling is by forced convection the main parameters are: (1) The temperature drop between the wall and the coolant; and (2) the velocity of the coolant. The coolant velocity is the unknown. The pressure drop through the cooling passage is finally determined and is largely a function of the coolant velocity (Equation [14]).[1]

The gas temperature, velocity, pressure, density, etc., vary considerably at different sections of the nozzle. They also vary with motor size, fuel-oxidizer ratio, and combustion pressure. Laborious calculation is avoided by rationalized methods of computations. One such method has been developed by the author's company (1).[2] Solution is mainly by graphical interpolation and simple calculation. The conditions in the cooling passage are similarly rationalized and a method for treating conditions in the wall is described herein and, in further detail, in (2). The process does not appear to need further radical improvement. Hereafter discussion is limited to the parameters involved.

Shortcomings of Calculations

The aforementioned method of calculating conditions on the gas side has been proved by experiment to give values of the correct order. However, discrepancies occur at the flame front (because of injector characteristics) and at the throat (where the flow is sonic and compressibility effects can be severe). These have been demonstrated with a type of apparatus in which different sections of the motor are separately cooled.

Since analytical evaluation of the foregoing deviations is not possible at present, the following approximations have been used:

1 The theoretical combustion temperature is reduced to allow for incomplete combustion, e.g.,

$$\frac{T_{B\ \text{actual}}}{T_{B\ \text{theoretical}}} = \left(\frac{C^*_{\ \text{actual}}}{C^*_{\ \text{theoretical}}}\right)^2$$

where

$$C^*_{\ \text{theoretical}} = \frac{gI_{sp}}{C_F}$$

and C^*_{actual} is obtained from test results,

$$C^*_{\ \text{actual}} = \frac{P_c \times A_t}{w} \times g$$

Numbers in parentheses refer to references at the end of the paper.

2 The temperature of the burning gases in the heat-transfer calculation is assumed to be equal to the "actual" combustion temperature up to the throat. This is sometimes referred to as the recovery temperature. Beyond the throat, free stream temperature is assumed. The same applies in the computation of radiant heat transfer.

Analysis based upon the aforementioned test results suffers from two major shortcomings:

1 The test method gives only an average value over the test section and will, therefore, be inaccurate in throat section.
2 No differentiation is made between radiation and convection heat transfer.

Fig. 2 shows a motor designed to give fairly accurate heat-transfer data. The whole motor is copper and is finned. The air gaps between fins prevent longitudinal heat flow except for the thin section forming the combustion-chamber wall. The radial heat transfer can then be determined by measuring the temperature at two known points in the wall. This is done with fine thermocouples.

By plating different sections with materials of known emissivity, and similarly isolating these sections with air gaps, the radiant heat transfer can be determined experimentally. The method may prove to be more reliable for highly luminous gases than for nonluminous gases since the effect of various wavelengths on the different monochromatic emissivities of the surfaces may give considerable errors.

Cooling-Passage Design

In practice the quantity of heat flow is determined primarily by conditions on the gas side. The problem, then, is to decide:

1 In what manner the coolant passage should be designed to handle this heat flow.

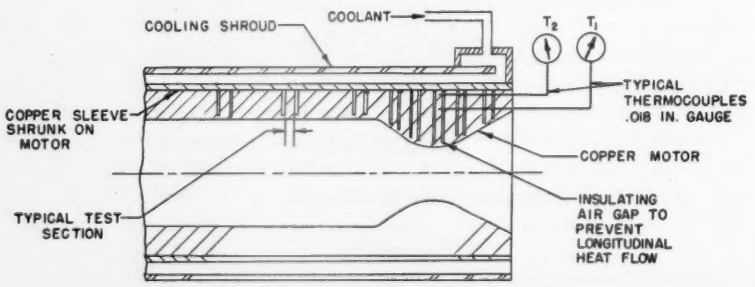


FIG. 2 TEST MOTOR FOR ACCURATE DETERMINATION OF HEAT-FLUX DENSITY

2 What material and wall thickness should be selected to handle the heat flow and maintain structural integrity.

The cooling process follows three main principles: (1) Forced convection; (2) film boiling; and (3) sweat cooling.

Forced Convection Cooling

By definition the wall temperature at the gas interface will be below the boiling temperature of the coolant; otherwise boiling occurs.

The coolant film coefficient (h_L) is determined from the equation:

$$\frac{q}{a} = h_L (T_W - T_L) \quad [1]$$

The velocity is then found from the well-known McAdams equation:

$$h_L = \frac{K}{4m}\left(\frac{4V_m\rho}{\mu}\right)^{8.0}\left(\frac{\mu C_p}{k}\right)^{0.4} \quad [2]$$

From the equation for conservation of mass the passage width is found. The pressure drop is next determined from the Fanning equation:

$$P = f\frac{L}{4m}\frac{\rho V^2}{2g} \quad [3]$$

Obviously, forced convection cooling will only be possible as long as there is sufficient temperature drop between the wall and the coolant. It can be achieved in rocket motors with coolants having low vapor pressures such as water, alcohol, aniline, etc. Even with these coolants the following phenomena sometimes occur near the throat where the heat transfer is generally highest:

1 In the case of small motors the cooling-passage width may be too small for fabrication.
2 In larger motors the high velocity may result in an excessive pressure drop.

Many propellants have vapor pressures high enough to prohibit forced convection in any part of the motor.

It is possible to increase the cooling-passage width by using spiral passages. Obviously since this increases the cooling-passage length it will increase the pressure drop, which becomes more critical as size increases.

Film Boiling Cooling

When it is not practical to maintain forced convection cooling, the wall temperature will increase until it reaches the boiling temperature of the coolant. There is a marked increase in heat transfer without change in coolant velocity and with little increase in wall temperature.

This sudden increase is due to the vaporization of an extremely thin film

of coolant at the wall interface. A breakdown in the cooling process is prevented by the condensation of this vapor into the coolant mass, which is at a lower temperature. This phenomenon is variously described in the literature including the Transactions of The American Society of Mechanical Engineers (3). In the past the designer has had no quantitative data upon which to design for boiling heat transfer and more satisfactory solutions to problems have been by trial and error. However, usable data at the high rates of heat transfer found in rocket motors are now being obtained with a type of apparatus in which the heat transfer is obtained by electrically heating a resistance tube. Various coolants at differing bulk temperatures, velocities, and pressures can be tested. Typical test results with water as the coolant were reported to the American Physical Society in June, 1948 (4). In addition to velocity, the vapor pressure, heat of vaporization, etc., will have a marked effect on the critical condition where vapor binding occurs.

Sweat Cooling

It was shown that when forced convection cooling is no longer possible a breakdown in the cooling process can be considerably delayed by the process of film-boiling. Eventually, however, the limit is reached where even this process becomes inadequate. The obvious and most practicable solution is then to reduce the rate of heat flow through the wall. There are two methods of accomplishing this: (1) By the infusion of an insulating layer of liquid or gas between the hot gases and the wall; and (2) by the insertion of a material which is a poor heat conductor.

The first method is known as sweat cooling. In this method the motor walls are porous so that the coolant will seep through as shown in Fig. 3. On reaching the combustion chamber the liquid vaporizes and is exhausted downstream. The adjacent combustion gases are cooled by the vaporization process. The heat flow through the wall is reduced because some of the heat transferred in the cooling process has been exhausted downstream. The phenomenon is of paramount value when using coolants having extremely high vapor pressures, such as liquid hydrogen. It also indicates that there may be no performance restriction due to cooling problems upon rocket motors using chemical energy.

Film cooling is a variation of sweat cooling. In one approach small radial or tangential holes are drilled in the motor wall allowing a portion of the coolant to pass through. It is used most often to reduce rates of heat transfer at the throat, for example, on large motor designs requiring thick walls for structural reasons, when walls of alloys of low conductivity are used, and also when coolants with medium vapor pressures such as nitric acid are used. Both the film and sweat cooling processes, however, do detract slightly from motor performance. In both instances it is possible that

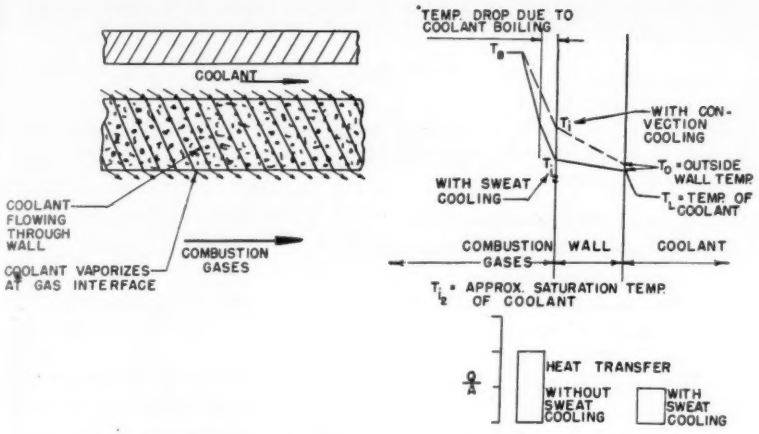


FIG. 3 DIAGRAM OF THE SWEAT-COOLING HEAT-TRANSFER PROCESS

the temperature through the entire wall can be kept down to the boiling temperature of the coolant (5).

Discussion of Cooling Problems

There are several shortcomings to the testing of boiling heat transfer at a constant rate of heat and coolant flow along an electrically heated tube as previously described. In a rocket-motor nozzle the heat-flux density, coolant velocity, passage diameter, shape, and width are constantly varying. It is logical to suggest electrically heating a small test rig simulating conditions in an actual motor. The conditions of heat transfer will be known and can be rigidly controlled.

It is suggested that all motors should be proof-tested to determine the efficiency of the cooling arrangement by running the motor at some pressure in excess of the working pressure to give a greater rate of heat transfer. The cooling passage would operate at the normal pressure.

Heat Flow Through the Wall

When cooling by forced convection was considered, it was suggested that a wall temperature slightly below the coolant boiling temperature should be selected. When cooling with film boiling is employed, the wall temperature will slightly exceed the coolant boiling temperature. The exact amount cannot be determined by calculation at present, but it is thought that 100 F is a good estimate of the average.

The simple equation for heat flow through a flat plate with constant thermal conductivity is:

$$\frac{q}{a} = \frac{K}{t}(T_i - T_o) \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots [4]$$

This equation is in error when considering the large radial heat flows through the cylindrical or spherical combustion chamber of a rocket motor. It makes no allowance for:

1 Longitudinal heat flow.
2 The reduction in heat-flux density as the surface area increases through the wall.
3 The variation of thermal conductivity with temperature through the wall.

Data on the variation of thermal conductivity of metals with temperature are surprisingly meager at the temperatures found in rocket-motor walls. A series of tests at Lehigh University have been sponsored to obtain accurate data on a variety of metals at the relevant temperatures.

The author's company has evolved a quick method for establishing the temperature gradient through a cylindrical or spherical wall of known material, once the heat-flux density is known. A report on this method will shortly be published. In calculating the heat flow through the wall an assumed inside wall temperature is used. It is then possible, by successive approximation, to obtain the exact inside wall temperature. Longitudinal heat flow is not considered to be important if the nozzles are made of low

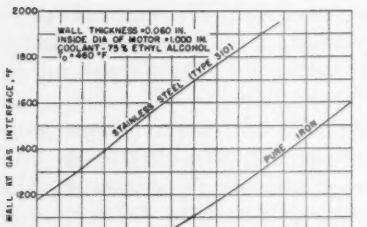

WALL THICKNESS = 0.060 IN.
INSIDE DIA OF MOTOR = 1.000 IN.
COOLANT - 75 % ETHYL ALCOHOL
T_o = 460 °F
STAINLESS STEEL (TYPE 310)
PURE IRON
ALUMINUM
COPPER
TEMPERATURE OF WALL AT GAS INTERFACE, °F
HEAT FLUX DENSITY AT GAS INTERFACE, BTU/SQ.FT.-SEC.

FIG. 4 RELATION BETWEEN WALL TEMPERATURE AND HEAT-FLUX DENSITY FOR SEVERAL MATERIALS

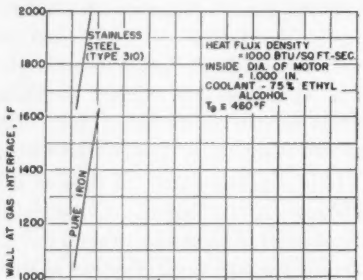

STAINLESS STEEL (TYPE 310)
HEAT FLUX DENSITY = 1000 BTU/SQ FT-SEC.
INSIDE DIA. OF MOTOR = 1.000 IN.
COOLANT - 75 % ETHYL ALCOHOL
T_o = 460 °F
PURE IRON
ALUMINUM
COPPER
TEMPERATURE OF WALL AT GAS INTERFACE, °F
WALL THICKNESS, INCHES

FIG. 5 RELATION BETWEEN WALL TEMPERATURE AND THICKNESS FOR SEVERAL MATERIALS

conductivity metals. With materials such as copper, however, the effect is highly important.

The heat-transfer conditions through the gas, chamber wall, and coolant are determinable. In practice, there are two additional important factors to be considered: (1) The thickness required for adequate strength; and (2) the maximum inside wall temperature that can be tolerated in the high-velocity corrosive gas stream.

The temperature problem will be examined first. The conditions are most critical at the throat. For convenience it will be assumed that there is film-boiling and the wall temperature at the coolant interface is 100 F above the boiling temperature of the coolant. Curves can then be drawn giving the relation between wall temperature at the gas interface and heat-flux density for various wall materials of constant thickness, Fig. 4. Curves can also be drawn showing the relation between wall temperature at the gas interface and wall thickness for a given heat-flux density, Fig. 5. Both these curves assume the coolant is a 75 per cent alcohol-water mixture at 350 psi.

The next step is to determine the maximum temperature that can be tolerated at the gas interface. There is little reliable information. The gases are flowing at high velocity and may be oxidizing or reducing in nature depending upon mixture ratio and the injector characteristics. A material which has a high melting point and good conductivity will, therefore, be evaluated on the temperature at which it oxidizes or otherwise loses its strength. It appears that the only logical way to evaluate materials is to test them in actual motors under controlled conditions. Fig. 6 illustrates a method of obtaining this data.

The rocket nozzle is an insert made of the material to be tested and is bolted to a standard 350-lb thrust combustion chamber. The cooling water is pressurized to 300 psi and flowed at 25–35 fps. This rate is considered sufficient to prevent failure from inadequate cooling. Measurement of the rate of cooling flow and temperature rise over the test section will give an accurate measurement of the heat rejection. The copper sleeve shrunk on to the test section serves as a baffle system to obtain the heat rise over the test section. Copper was selected to avoid local hot spots which might occur if the baffles were part of the nozzle. The wall temperature can be measured by drilling 0.018-in. longitu-

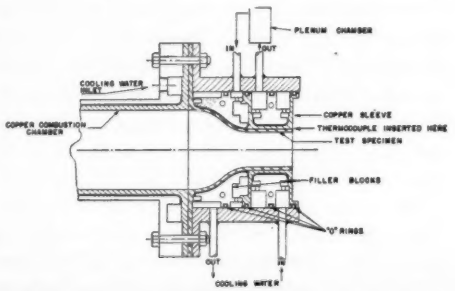


FIG. 6 SKETCH OF MATERIALS TEST ROCKET MOTOR

dinal holes at the interface to accept thermocouples without fear of local overheating. An accurate estimate of the wall temperature at the gas interface can then be obtained from (2).

It is proposed to vary the wall thickness of the various test materials so that the assumed maximum inside wall temperature at which failure would occur is obtained when the rate of heat transfer is 1000 Btu per sq ft per sec (corresponding roughly to a chamber pressure of 300 psi with alcohol-liquid-oxygen propellants).

The method of testing will then be to increase the chamber pressure and vary the propellant-mixture ratio until the desired inside wall temperature and flame condition are obtained, or until failure occurs. Thermocouple data should indicate if failure was caused by a breakdown in the cooling process. The value of protective plating can be checked and various propellant combinations can be tested.

Strength

A more complete but still inadequate discussion of the strength problem will be found in BAC Meteor Report No. 13 (6).

Three conditions must be satisfied:

1 The "cold" start, with a high differential collapsing pressure.
2 The explosive start, with a bursting pressure.
3 The "hot" running condition.

Failure in case 1 can be estimated from the formula for instability of thin shells and checked by test.

Failure in case 2 can be checked simply by detonating a given quantity of propellants in the chamber.

Failure in case 3 can be checked by running tests. The motor is operated at a chamber pressure exceeding the working pressure. This will slightly overheat and weaken the wall. The pressure in the cooling passage will then be increased to some predetermined value in excess of the working pressure. A separate cooling system will be necessary to accomplish this.

In the hot case the strength and modulus of elasticity of the material may be markedly reduced. The problem is further complicated by the temperature gradient through the wall which causes hoop-compression stresses on the inner wall and hoop-tension stresses on the outer wall. Calculation will show that if the theoretical stresses were realized, failure would occur. In practice, it is thought that the material on the inner wall yields and relieves the stress on the outer wall.

In this respect, testing with fire tubes at different pressures might produce useful data but it is an inconvenient, laborious, and expensive method of testing. For purposes of research it might be possible to test tubes made of low-conductivity material with temperature elastic and plastic properties similar to metals. The volume and temperature of the hot gases might be

kept to very low values and, with bakelite-base materials and using polarized light, the stress pattern could be studied.

Materials Selection

For illustrative purposes materials will be divided into three main types. However, at present, the merit rating of metals is difficult to define.

High-Conductivity Materials: The most popular varieties are copper and aluminum. Their high thermal conductivity places little restriction upon wall thickness when used in rocket motors, reduces tendency to burn out due to local hot spots, and they have no temperature-differential stress problem.

Aluminum has good corrosion resistance. Its shortcomings arise from its low softening and melting temperatures. The excellency of copper suffers from poor corrosion characteristics with some common oxidizers, cost, strategic shortage, and poor strength-weight ratio.

Medium-Conductivity Materials: These are usually carbon steels or pure metals such as iron and nickel. The use of nickel is restricted by its cost and shortage. Its corrosion resistance is variable, machining qualities average, and temperature characteristics good. Iron oxidizes at low temperatures but steel is a little better and is a good all-round material. It plates easily and its further development may be on these lines. The melting point of all these metals is high.

Low-Conductivity Materials: In this category fall the highly alloyed materials, notable for their ability to retain strength at high temperature and resistance to oxidation. The poor conductivity resulting from the alloying detracts from the high-temperature strength qualities and places a severe restriction on wall thickness. The strength-weight ratio will be excellent, the resistance to corrosion and oxidation excellent. Cost, availability, welding, machining, and forming qualities are all generally poor. There is little demand in rocket motors for the high-temperature creep-resistance materials developed for gas turbines.

In general, it appears that in the selection of material for construction of rocket-motor combustion chambers, carbon steels offer the most advantages. Alloys of iron, cobalt, chromium, etc., may be used to overcome interim corrosion problems but long-term development may result in steels again being favored.

Conclusions

In conclusion, it can be stated that the problems of heat transfer and cooling design in rocket motors can be evaluated by orthodox and simple test methods. Analytical prediction can be made with semiempirical methods.

Cooling problems, though far more severe than found in general engineering, are capable of satisfactory solution. There need be no restriction upon rocket-motor performance using chemical energy because of cooling problems.

Motors constructed from common metals can be made to function satisfactorily. Improved materials will need to be cheap and have high thermal conductivity, high-temperature strength, high-oxidation temperature. Present available high-temperature materials are unsatisfactory generally because of their low thermal conductivity.

The art of cooling rocket motors can be reduced to science. No new engineering principles are required. Considerable improvement will be possible when laboratory testing catches up with the rapid advance made by trial and error design.

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2 "A Study of Heat Flow Through Metals with Variable Conductivity," by Edmund Ellion, Bell Aircraft Corporation Meteor Report No. BAC-21 (not yet published).

3 "Heat Transfer to Water Boiling Under Pressure," by E. A. Farber and R. L. Scorah, Transactions of The American Society of Mechanical Engineers, vol. 70, no. 4, May, 1948.

4 "Heat Transfer to Water at High Heat-Flux Densities With and Without Surface Boiling," by F. Kreith and M. Summerfield, Heat Transfer and Fluid Mechanics Institute of American Physical Society, June 22 meeting, 1948.

5 "Experimental Study of Cooling by Injection of a Fluid Through a Porous Material," by Pol Duwez and H. L. Wheeler, Jr., Journal of Aeronautical Sciences, vol. 15, no. 9, September, 1948.

6 "Requirements for Metals for Regeneratively Cooled Rocket Motors," by J. L. B. Selwood, Bell Aircraft Corporation Meteor Report No. BAC-13, July 1, 1947.

ARS PREPRINTS AVAILABLE

The following preprints are available at 25 cents each from the secretary, American Rocket Society, 29 West 39th Street, New York 18, N. Y.

"A Survey of Injector Designs for Use in Liquid Propellant Rocket Motors," by B. N. Abramson, Bell Aircraft Corporation, Buffalo, N. Y.

"Rocket Engine Development," by Roy Healy, M. W. Kellogg Company, Jersey City, N. J.

"Creation of Rocket Propellant Gas Source for Rocket Motor Propellant," by C. J. Turansky and R. D. Rinehart, Bell Aircraft Corporation, Buffalo, N. Y.

"An Investigation of Some of the Parameters Affecting Over-All Rocket Performance," by C. H. Harry, Glenn L. Martin Company, Baltimore, Md.

"The Air Force Experimental Rocket Engine Test Station," by K. F. Mundt, Aerojet Engineering Corporation, Azusa, Calif.

THE PROSPECTS OF JET-REACTION FLIGHT (PART I)[1]

By Eugen Saenger

Consulting Engineer, Paris, France

Symbols

a = sonic speed (m/sec)
c = effective efflux velocity (m/sec)
c_a = lift coefficient (dimensionless)
c_m = moment coefficient (dimensionless)
c_s = thrust coefficient (dimensionless)
c_p = specific heat at constant pressure (kcal/kg°)
c_H = most probable molecular velocity before collision (m/sec)
c_R = most probable molecular velocity after collision (m/sec)
d = diameter, wing thickness (m)
f = rocket cross-sectional areas (m²)
f' = smallest nozzle cross section (m²)
g = acceleration of gravity (m/sec²)
c_w = drag coefficient (dimensionless)
i_{pH} = impulse before collision (kg/m²)
i_{pR} = impulse after collision (kg/m²)
i_τ = impulse parallel to wall (kg/m²)
k = coefficient (dimensionless)
l = length of flow (m)
m = mass (kg sec²/m)
p = pressure (kg/m²)
p_m = exit pressure (kg/m²)
p_o = pressure in combustion chamber (kg/m²)
q = dynamic pressure $v^2/2$ (kg/m²)
s = trajectory (m)
t = time (sec) wing chord (m)
t_f = fire edge temperature (deg C)
t_k = cool edge temperature (deg C)
v = flying speed (m/sec)
v_1 = initial speed in gliding flight (m/sec)
w = velocity of flow (m/sec)
A = mechanical equivalent of heat (kcal/mkg)
F = area, wing area (m²)
F_2 = frontal area (m²)
G = weight (kg)
G_0 = original weight (kg)
H = flight altitude (m)
H_u = lower calorific value (kcal/kg)
K = force (kg)
L = output (mkg/sec)
M = molecular weight (dimensionless)
P = thrust (kg)
P' = free thrust (kg)
Q = quantity of heat (kcal/sec)
R = radius of earth (m), individual gas constant (m/°)
$\Re$ = general gas constant (m/°)
Re = Reynolds number (dimensionless)
T = absolute temperature (deg K)
V_0 = combustion chamber volume (m³)
W = drag (kg)
Z = number of collisions of molecules (dimensionless)
α = angle of incidence (°), heat transfer coefficient (kcal/m²h°)
α_x = local angle of incidence (°)
α_{st} = angle of incidence of compression shock (°)
α_z = local angle of incidence of a fore-positioned surface element (°)
Δ = finite change (dimensionless)
ϵ = lift/drag ratio c_w/c_a (dimensionless)
λ = thermal conductivity (kcal/mh°)
σ = high-temperature stability (kg/m²)
τ = frictional tension (kg/m²)
ρ = density (kg sec²/m⁴)
η = efficiency (dimensionless)
κ = polytrope adiabatic exponent (dimensionless)

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[1] Reprinted from *Inter Avia*, vol. III, September, 1948, pages 499–506. Dr. Saenger's paper was published in three parts. Parts II and III will be reprinted in the June and September issues, respectively, of the ARS JOURNAL.

General

LASTING among new technological developments of World War II are notably atomic energy, for which most credit goes to the U.S.A.; radar, a primarily British contribution; and jet reaction, a mode of propulsion fostered originally by Germany.

Throughout almost 40 years, aircraft have been developed exclusively in conjunction with reciprocating engines. Dictated by the requirements of the last war, there suddenly arose a whole series of new prime movers for aircraft, such as resonance ducts, turbojet power plants, rocket engines, and aerothermodynamic ducts (athodyds or ramjets), which enables us to speak of a new branch of aeronautical engineering: aircraft propulsion by thermal jet reaction.

The understandable interest, from the scientific, technical, commercial, military, political, and general public standpoints, evinced by the wide vistas lying before this new mode of flight propulsion justifies an attempt to assess it. However, such an assessment must necessarily remain a strictly personal venture, even though the aeronautical scientist, called upon to offer his views, is able to base himself on a large number of real facts and conditions, particularly physical ones, which will in all probability find practical application and, thus, point out to the engineer the way ahead.

The aim in air transport, as in every technical branch of transport, is the carriage of given loads over predetermined routes. A primary implication in connection with jet-reaction flight technique concerns an increase in the transport speed together with a maximum restriction, or even a reduction, in the operating costs. Flying greater distances and at greater altitudes, thus bringing aircraft further away from the surface of the globe, is only a secondary implication.

The first means of propulsion from this series of new aircraft prime movers to be put into practical application was the resonance duct, which was unable to fulfill its designer's hopes as regards speed. Even for the future, its prospects are only moderate. Nevertheless, it succeeded under special transport conditions in reducing the operating costs to such an extent that it will not disappear completely without further ado.

The second new type of power plant to find general introduction was the turbojet engine. Although it did not reduce the operating costs, it offered a wider range of operational flying speeds, within the range of about 700 to 1000 km/h. Further increases in speed, even to a piercing of the sonic barrier, may be anticipated once the present airframe limitations have been overcome. Meanwhile, its operational speed will be kept within the above limits referred to, since more suitable propulsion systems are available for higher flying speeds, which are still in the experimental stage owing to the lag in airframe development: ramjet engines and rocket power plants.

Chemical fuels will definitely remain in the lead during the next decade of aeronautical development. Their earlier replacement by atomic energy is hardly to be imagined.

The consumption of these classical fuels presents a determinant factor for both the operating costs and the range of aircraft, thus leaving the concept of engine efficiency as an important index, even though the different fuel bases of modern aviation engines and the influences of constructional costs and absolute output have already narrowed down this concept's earlier cardinal importance.

Fig. 1 reveals the unique superiority of the piston engine with propeller throughout almost the entire subsonic speed range, the immense importance of the ramjet within the range of Mach 1 to 4, and the exclusivity of pure rocket propulsion for still faster aircraft. The separate peak efficiencies are particularly prominent. Worthy of note is the pronounced efficiency gap in the vicinity of sonic speed ($\sim$1000 km/h), about where the turbojet engine begins to find application. The latter's field of application is not limited to the subsonic range, but its efficiency at higher supersonic speeds naturally goes over toward the more simple ramjet engine as soon as the ram compression of the latter becomes comparable with the compression effect of turbojet engines. Turbojet engines are therefore indicated for flying speeds around Mach 1, in which domain, however, many aircraft reveal unfavorable polar curves and, consequently, short ranges.

Most noticeable from the curve pertaining to the resonance duct is that the efficiency values are low throughout. The eventual economic superiority of this engine is, therefore, solely attributable to its low constructional costs, a circumstance which confronts us again when considering the subsonic ramjet engine. Though this has oddly escaped practical engineering so far, it may lead to a replacement of the resonance duct by the subsonic ramjet engine for flight at high subsonic speeds.

Thus it may be seen that, for the outgoing era of chemical fuels in aeronautics, two systems of aircraft propulsion are coupled with the main prospects of jet-reaction flight: ramjet engines and rocket power plants.

Ramjet Propulsion

Working Principle: Known for 35 years, thanks to the proposals of Lorin, the principle of this engine mainly consists of designing a duct in such a way that air continuously streaming in against flight direction is first decelerated and compressed, then heated, and finally ejected at accelerated speed, whereby the entire procedure is continuous and requires no rotating part.

The revolutionary technical significance of this principle of propulsion is that it no longer calls for special engine nacelles which, today, are developed separately from the aircraft and attached or built into the finished airframe, but that it rather represents a design principle for fuselages or wings in

order to make the latter's drag negative so that the power plant and cell, and also the aerodynamic forces due to drag, lift, and propulsion of the aircraft, combine with one another and obey common aerodynamic laws.

Thrust Coefficient versus Mach Number: If the frontal area F_2 of the engine is sufficiently large in comparison with the cross-sectional area F_1 of the air intake, then the vector sum of all gas-pressure differences as compared with the undisturbed ambient air pressure p_1 at the engine walls (as long as they are caused by the air masses flowing through), thus the "thrust P," may be expressed with good approximation by the formula:

$$P = \rho_1 F_1 v^2 \left[\sqrt{T_3/T_2} \sqrt{\frac{1 - \left(\frac{p_1}{p_1 + \Delta p}\right)^{(\kappa-1)/\kappa}}{1 - \left(\frac{p_1}{p_1 + \Delta p/\eta_D}\right)^{(\kappa-1)/\kappa}}} - 1 \right] \quad \ldots\ldots [1]$$

wherein ρ_1 signifies the density of the undisturbed air and v its speed, T_3 the absolute ram temperature of the in-flowing medium, and T_2 that of the out-flowing medium, thus $(T_3 - T_2)$ the amount by which it is heated; Δp the pressure increase really attained in the diffuser, and $\Delta p/\eta_D$ the pressure increase attainable through isentropic compression.

The diffuser efficiency η_D at subsonic speeds is always close to unity, but at supersonic speeds this only arises in a case of isentropic compression.

For supersonic diffusers with perpendicular compression shock at entry, the following formula is approximate:

$$\eta_D = \frac{\Delta p}{\Delta p_{is}} = \frac{\frac{4}{\kappa+1}(1 - a^2/v^2) + \frac{(\kappa - 1 + 2a^2/v^2)[9\kappa - 1 + (6 - 4\kappa)a^2/v^2]}{[2\kappa - (\kappa - 1)a^2v^2]4(\kappa + 1)}}{\frac{\left[\frac{\kappa-1}{2} v^2/a^2 + 1\right]^{\frac{\kappa}{\kappa-1}} - 1}{\frac{\kappa}{2} v^2/a^2}} \quad \ldots [2]$$

In the case of isentropic compression, thus especially at subsonic speeds, the thrust formula may be expressed more simply:

$$P = \rho_1 F_1 v^2 (\sqrt{T_3/T_2} - 1) \ldots\ldots [3]$$

At subsonic speeds, if profile diffusers are used, the gas outlet cross section F_4 can be fixed and F_1 can be variable, thus yielding a simpler formula:

$$P \approx \rho_1 F_4 v^2 (1 - \sqrt{T_2/T_3}) \ldots\ldots [4]$$

Condition:

$$\frac{F_4}{F_1} \approx \sqrt{T_3/T_2} \ldots\ldots [5]$$

The engine efficiency is simply:

$$\eta = \frac{APv}{Q} \ldots\ldots [6]$$

if $Q = \rho_1 F_1 v g c_p (T_3 - T_2)$ is the quantity of heat transmitted and A the mechanical equivalent of heat.

For the thrust forces, we can define a thrust-force coefficient $c_s = P/qF_2$ in the same manner as for the other air forces, and find that the curve of these thrust coefficients versus the Mach number is similar to the well-known curve of the air-force coefficients, as for example Fig. 1 shows, i.e., at subsonic speeds, air forces which increase, practically, with the square of flying speed, and later slower, only decreasing once considerable Mach numbers are attained.

Correspondingly, the entire engine efficiency also increases, initially with the square of the flying speed,

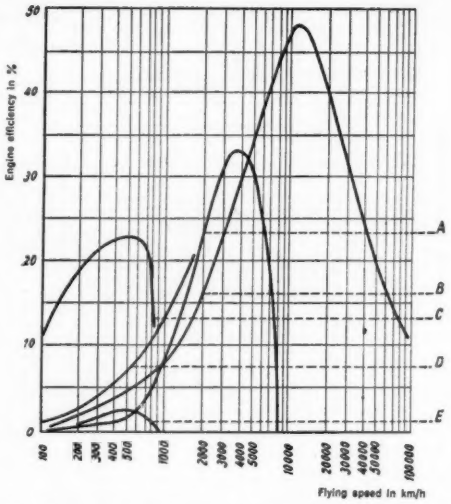

FIG. 1 ENGINE EFFICIENCY VERSUS FLYING SPEED (E = Resonance duct; D = Turbojet engine; C = Piston engine with propeller; A = Athodyd, ΔT = 1000 C; B = Oil-oxygen rocket, c = 3200 meters per sec)

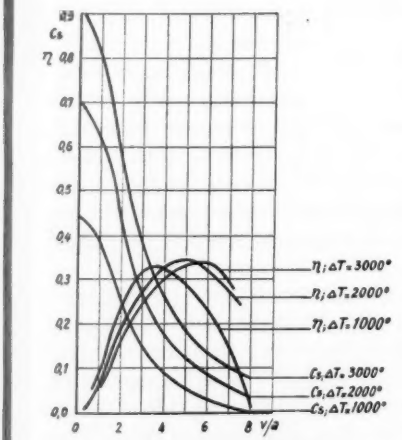

FIG. 2 THRUST COEFFICIENTS c_s AND TOTAL EFFICIENCIES η OF ATHODYD POWER PLANT FLYING IN STRATOSPHERE, WITH AIR HEATED THROUGH ΔT = 1000, 2000, AND 3000 C. CONDITION: PERPENDICULAR COMPRESSION SHOCK AT DIFFUSER ENTRY FOR F_1/F_2 = 0.16

and attains a maximum at high Mach numbers, this maximum being dependent on the magnitude of the temperature increase and also on the degree to which compression in the engine diffuser has been achieved isentropically.

Engine Diffusers: With F_1/F_2 = 0.16, Fig. 2 shows the conditions for the practical limit case with perpendicular compression shock at the diffuser entry.

This limit case shows lower efficiencies than in the case of isentropic compression but leads up to an especially simple constructional design of engine diffuser, which is suitable without adjustment for the entire Mach range, and which is recognizable in Figs. 3 and 5, where an experiment on a flying test stand is presented, showing

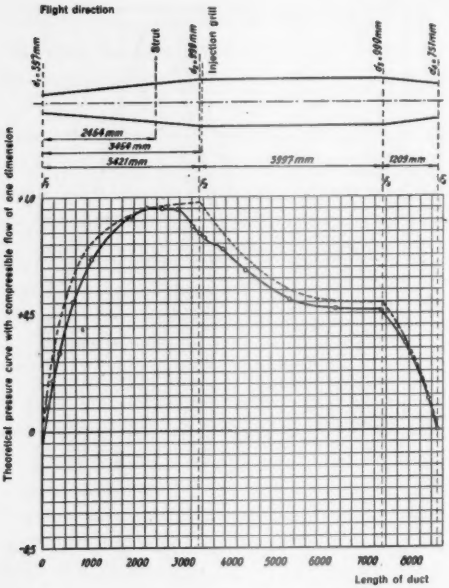

FIG. 3 CURVE MEASURED FROM RATIO OF INTERIOR POSITIVE PRESSURE Δp TO DYNAMIC PRESSURE q_{kompr} VERSUS LENGTH OF LORIN DUCT, AT 4 KM ALTITUDE WITH FULL POWER APPLIED (FULL-LINE CURVE)

($H = 4$ km; $q = 546$ kg/m²; $v/a = 0.352$; $p = 6075$ kg/m²; $T_0 = 265$ K; $c_s = 0.55$)

the results in the form of the pressure distribution measured in a rotationally symmetrical engine by J. Bredt.

Alongside this type of classical diffuser, in the form of a truncated cone, studies were also conducted at supersonic speeds with the oblique-shock diffusers designed by Oswatitsch, which separate the individual perpendicular compression shocks into a multitude of oblique shocks and thereby attain less entropy increase. At subsonic speeds, the profile diffusers of Ruden were found to be advantageous, particularly as regards more favorable possibilities of regulating the engine.

Whereas straight diffusers create engine thrusts by means of positive pressure at the interior walls, the profile diffusers do this by means of negative pressure; thus local accelerations at the diffuser nose. Owing to this, the application of profile diffusers becomes all the more restricted as sonic speed is reached, as can be seen from a case presented in Fig. 4 where an exceeding of the local sonic speed at the diffuser nose is not permissible.

Heating with Oil. For heating ramjet engines with liquid fuel, the injection grill shown in Fig. 5 has proved its utility in flight tests. The regular distribution of a large number of injection nozzles over the entire flow cross section enables good macromixture to be effected; injection against the direction of flow increases the relative speed and the storage life of the fuel in the combustion chamber, thereby yielding good mixture preparation; finally, the utilization of twist-injection nozzles enables micromixture to be achieved.

This arrangement enables exceptionally complete combustion to be achieved at altitudes up to 7000 meters, even with only moderate injection pressures and without prevaporization of the fuel. The behavior of the

turbulent liquid-fuel combustion at lower air pressures presents a special research problem, concerning which only partial results, though positive in character, have been attained so far.

Heating with Coal: As regards heating ramjet engines with liquid coal, it seems that no data gained from experience are available, although this method might be of commercial significance.

On the other hand, extensive data are available on heating with solid coal, showing in the main that the speed and speed change of solid-coal combustion can at best satisfy the requirements of horizontal flight at steady subsonic speed, and could not be used for fast ascents or supersonic flight.

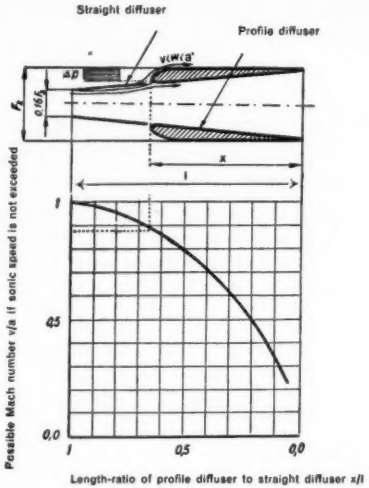

FIG. 4 PERMISSIBLE APPROACH OF FLYING SPEED TO SONIC SPEED, USING A PART-PROFILE DIFFUSER INSTEAD OF A STRAIGHT DIFFUSER, IF THE LOCAL SPEED OF FLOW AROUND THE DIFFUSER NOSE DOES NOT EXCEED SONIC SPEED

Heating with Atomic Energy: In principle, the ramjet engine is well suited to the utilization of atomic heating in the form of an atomic radiation heating, which exploits the high absorption capacity of the air for α and β particles. One could imagine the distribution of the radiation sources over a grill similar to the distribution of injection nozzles presented in Fig. 5, so that the range of a few centimeters, as possessed by these nuclear particles in air compressed as it is here, enables the flow section to be covered evenly.

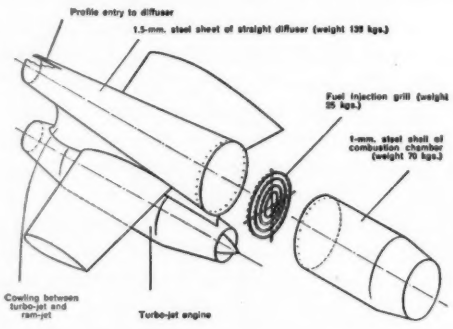
Profile entry to diffuser
1.5-mm. steel sheet of straight diffuser (weight 138 kgs.)
Fuel injection grill (weight 25 kgs.)
1-mm. steel shell of combustion chamber (weight 70 kgs.)
Cowling between turbo-jet and ram-jet
Turbo-jet engine

FIG. 5 MAIN COMPONENTS OF AN EXPERIMENTAL LORIN ENGINE WITH OIL HEATING, BUILT ON TO A MESSERSCHMITT ME 262

Combustion Chambers: The third important element of the ramjet engine, alongside the diffuser and the heating, is the combustion chambers. Instead of the originally used constant-pressure combustion chambers, whose cross sections increased proportionally to the heating of the air, preference is now given to cylindrical combustion chambers of the type shown in Figs.

3 and 5, in which the gas pressure during the heating process, particularly at subsonic flying speeds, drops noticeably.

Fire Walls: The usually high gas temperatures make it necessary to build combustion chambers of thin-walled metal shells, whose outer edges are exposed to the relative wind and thereby cooled very thoroughly.

The critical thermal requirements in athodyds concern the region lying downstream of the end of the diffuser, and differ according to whether the flight is at subsonic or supersonic speed.

At subsonic speeds, it has been seen from flight experience that, in the case of stoichiometric combustion of hydrocarbons, at a temperature of 2400 K, wall temperatures up to 600 C occur roughly one meter behind the injection grill at flying speed of 100 meters per sec, while the wall zones situated farther aft are somewhat cooler. This temperature maximum is attributable to radiation from the reaction zone exceeding the gas convection and normal gas radiation. This peak temperature would probably become less apparent at higher flying speeds, because the heat loss by convection increases with the flying speed faster than the reacting luminescence.

Under these conditions it is to be assumed that, at high subsonic speeds, air temperatures up to 4000 K obtained by nuclear reaction could be used without endangering engine walls of high quality steel.

The thermal requirements are considerably stricter at supersonic speeds. The compression temperature in the diffuser increases by about $v^2/2000$ C, where v is the flying speed in meters per sec. The temperature increase due to friction at the duct's outer walls amounts to about 85 per cent of the ram temperature, thus, for instance, causing a boundary-layer temperature of 1500 K at a Mach number of $v/a = 5.9$, which must also be regarded as the upper limit of the wall temperature of the ramjet; i.e., at this flying speed the cooling effect of the relative wind has completely disappeared and the wall may not be further heated from the interior. With heat-resistant steel as constructional material and without special cooling processes, the flying speed of the ramjet engine is therefore absolutely limited, for thermal reasons, at Mach number mentioned. In practice the limit is even lower.

Fig. 6 shows the compression and frictional temperature plotted against the Mach number. In the distance of 2060 C, and lower—owing to the increasing specific heats of the gases—less above the compression temperature curve, runs the curve of the flame temperatures of stoichiometric combustion of gasoline. With the aid of the usual gas-convection calculation there may be plotted, in approximation, a wall-temperature curve which lies between the curves of the combustion temperature and the frictional temperature, and cuts the temperature limit of heat and oxidation-resistant steels of 1500 K, at about $v/a = 3.5$. Higher flying speeds under these conditions, with a ramjet engine functioning at full power with gasoline, are not possible for thermal reasons.

Every other type of heating to the same temperature yields practically the same limit speed. Atomic heating to higher temperatures, of course, presents correspondingly lower limits.

An extension of this limit by using lower combustion temperatures, as shown in Fig. 6, is only possible as long as the then decreasing thrust coefficient suffices to overcome the drag coefficient of the engine moving at steady speed, and then only within very narrow limits.

During all attempts to raise the speed limit of the engine, i.e., by application of tungsten-carbide or titanium-carbide for the fire walls, it must furthermore be borne in mind that the airframe's walls are also under the influence of frictional heat. As Fig. 6 shows, this must lead to serious difficulties at high Mach numbers.

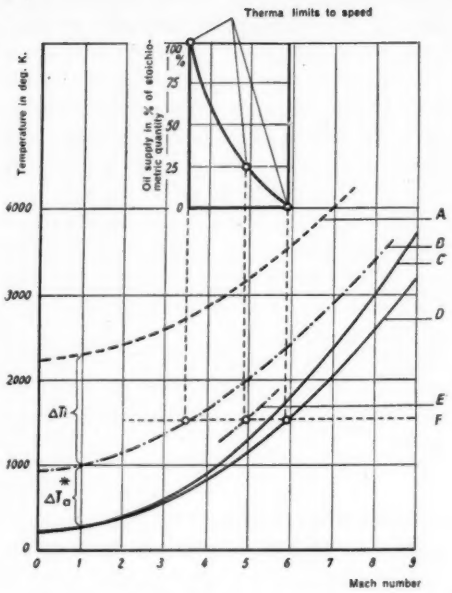

FIG. 6 LIMITS OF SAFE FLYING SPEED OF ATHODYD IN VIEW OF THERMAL REQUIREMENTS

A = Flame temperature T_3 stoichiometric
B = Calculated wall temperature in a case of stoichiometric combustion of oil
E = One-fourth stoichiometric combustion of oil
C = Compression temperature $T_2 = T_1 + (v^2/2000)$
D = Frictional temperature $T_1 + (v^2/2360)$
F = Temperature limit of high-temperature-stable steels
* Wall-temperature measurement

For thermal reasons, therefore, the flying speed of ramjet engines seems to find a limit at around Mach 3 to 4.

Rocket Engines

Effective Efflux Velocity: The range in unpowered gliding flight of a supersonic aircraft, with lift-drag ratio ϵ and initial flying speed v_1, may be expressed in good approximation by the relation:

$$s = -\frac{R}{2\epsilon}\ln(1 - v_1^2/Rg) \approx \frac{v_1^2}{2\epsilon g}\left(1 + \frac{v_1^2}{2Rg} + \frac{v_1^4}{3R^2g^2} + \frac{v_1^6}{4R^3g^3} + \dots\right) \quad [7]$$

whereby R signifies the radius of the earth (in meters). With $v_1 = 1120$ meters per sec, the influence of the second term becomes greater than one per cent of the first term; with $v_1 = 3290$ meters per sec, the influence of the

third term becomes more than one per cent greater than that of the first, while the second term is already 8.6 per cent of the first. At $v_1 = 7900$ meters per sec, the range, in view of the influence of the earth's curvature, becomes equal to infinity, instead of 20,000 km per sec, which the usual first term, not taking into acount the supporting effect of the centrifugal force due to the curved trajectory of the aircraft following the surface of the earth, would furnish with $1/\epsilon = 6.4$.

Thus the range first increases as the second power and later as a quickly increasing power of the initial speed v_1, to become almost infinite at a value of v_1 of over 7900 meters per sec; this denotes that the trajectory takes on the character of a planetary path about the earth.

With a single-stage rocket engine, in which the weight drops from an original value G_0 to an end value G_1, and the ratio of the rocket thrust to the original weight is high, and for which the effective efflux velocity is c, a powered end-flight speed, thus initial speed in gliding flight, v, of the magnitude:

$$v_1 = kc \ln (G_0/G_1) \quad [8]$$

may be obtained, whereby $k < 1$ is a coefficient which takes into account the partial absorption of the thrust by air drag and by weight components lying at a tangent to the trajectory, and which itself only slightly depends on v_1 in the sense that it approaches unity as v_1 increases, and under favorably selected practical conditions, on the average can be estimated at about $k = 0.8$–0.9.

Consequently, the dependence of the range s on the effective efflux velocity c, for single-stage rocket engines, yields the formula:

$$s = -\frac{R}{2\epsilon} \ln \left[1 - \frac{(kc)^2}{Rg} \ln^2 G_0/G_1\right] \approx \frac{(kc)^2 \ln^2 G_0/G_1}{2\epsilon g}\left(1 + \frac{(kc)^2 \ln^2 G_0/G_1}{2Rg} + \frac{(kc)^4 \ln^4 G_0/G_1}{3R^2g^2} + \ldots\right) \quad [9]$$

The range therefore begins by increasing as the second power, and later increases as a higher power of the effective efflux velocity, so that the latter's influence on the range is most apparent, being greater than that of the lift-drag ratio and for the most part greater than that of the mass ratio G_0/G_1; this circumstance is also evident from Fig. 7.

This justifies an effort to increase the effective efflux velocity at almost any price and, under given circumstances, even at the penalty of G_0/G_1 or ϵ.

As regards the definition of this highly important rocket index, it is odd that a wide divergency of opinion still prevails.

We can, for instance, calculate this with an unvarying effective efflux velocity referred to absolute vacuum, which possesses the following characteristics.

According to Fig. 8, in a rocket-driven flying body, the unvarying effective thrust P is equal to the vector sum of all absolute gas pressures on the

surfaces touched by the flame; thus, the inner walls of the rocket, and the drag W is equal to the vector sum of all absolute pressures and frictional tensions on the outer surfaces exposed to the wind. The force available for accelerated or ascending flight is $K = P - W$.

On the test stand, or in the aircraft, a thrust dynamometer is used to measure the available thrust $P' = P - p_a f_m$, whereby p_a is the atmospheric pressure.

In the wind tunnel or in free flight, the smaller drag $W' = W - p' f_m$ is measured on the unpowered aircraft, whereby p' is the absolute air pressure downstream of the nozzle cross section f_m. Hence these two forces cannot be used for equilibrium of the aircraft until a correction has been made: $K = P - W = P' - W' + f_m(p_a - p')$. It would consequently not be practical to derive the effective efflux velocity, as below, from the thrust P':

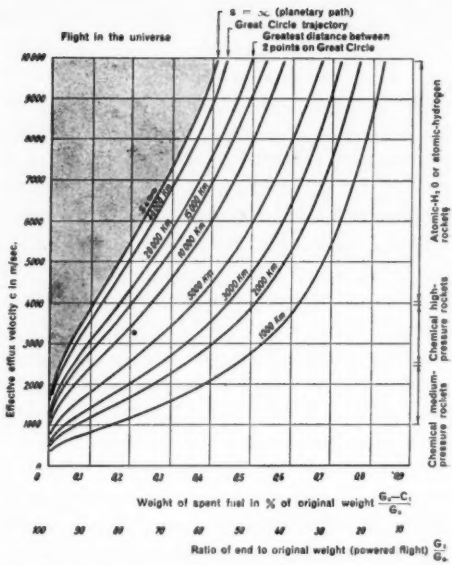
s = ∞ (planetary path)
Great Circle trajectory
Greatest distance between 2 points on Great Circle
Flight in the universe
Effective efflux velocity c in m/sec.
Atomic-H, O or atomic-hydrogen rockets
Chemical high-pressure rockets
Chemical medium-pressure rockets
Weight of spent fuel in % of original weight
Ratio of end to original weight (powered flight)

FIG. 7 GLIDE TRAJECTORIES s OF ROCKET AIRCRAFT, WITH SINGLE-STAGE ROCKET ENGINES, VERSUS FUEL PORTION OF ORIGINAL WEIGHT AND VERSUS EFFECTIVE EFFLUX VELOCITY c, FOR A LIFT-DRAG RATIO $1/\epsilon = 6.4$ AND A MEAN EXPLOITATION OF 90 PER CENT OF THRUST FOR ACCELERATION

$$C' = \frac{P'}{dm/dt} = C_m + \frac{(p_m - p_a)f_m}{dm/dt} \quad \ldots\ldots\ldots\ldots\ldots\ldots [10]$$

this also for the reason that this value is no engine index, but depends on the surrounding atmosphere, the air pressure and hence the weather, the altitude, and so forth. Preference is much rather given to its derivation from the unvarying effective thrust P:

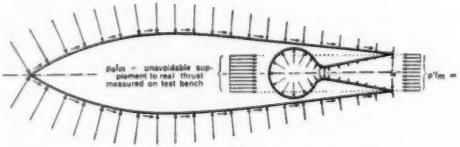

FIG. 8 INTERRELATION OF DRAG AND THRUST WHEN AIRCRAFT IS IN FLIGHT, WITH CORRESPONDING VALUES MEASURED ON THE GROUND ($p'f_m$ = Unavoidable supplement to real drag measured in wind tunnel)

$$c = \frac{P}{dm/dt} = c_m + \frac{p_m f_m}{dm/dt} \quad \ldots\ldots\ldots\ldots\ldots\ldots [11]$$

Meanwhile, however, this process involves a transfer of the inconvenient force $p_a f_m$ to the aerodynamical side, where it is likewise unfavorably placed.

The numerical values of c are always superior to those of the speed of flow c_m of the gases in the nozzle outlet; in ordinary liquid-fuel rockets, the difference approaches 14 per cent.

Rocket Fuels: The most important means of attaining high efflux velocities consists in the choice of suitable fuels of high calorific value, the utilization of high combustion pressures, and the best suited design of combustion chamber, nozzle, and fuel feed system.

During the first development stage of rocket-flight technique, the choice of fuels was cautiously restricted to compounds of low calorific value; for example, dilute alcohols (methanol), amines (hydrazine hydrate, aniline), vinyl ethers, etc., and as oxygen carriers, hydrogen peroxide, nitric acid, and, only exceptionally, liquid oxygen, were chosen; whereby sometimes quite incidental characteristics, such as the behavior of the fuels as coolants, as boosters, etc., played a part in the selection and the combustion temperatures were kept below 3000 K, while the effective efflux velocity was little more than 2000 meters per sec.

The present development stage, from the fuel standpoint, is centered on hydrogen compounds of high calorific value (hydrocarbons, gas oil, hydrogen), and liquid oxygen (eventually enriched with liquid ozone), while for special functions use was made of special materials, such as injected diethyl zinc for ignition, water, or mercury, etc., for cooling. The combustion temperatures are in the vicinity of 4000 K, and the efflux velocity between 3000 and 4000 meters per sec.

In a later, third stage, development will surely involve the heating with atomic energy of masses of low molecular weight, such as water or hydrogen, so that high efflux velocities can be anticipated at limited combustion temperatures, e.g., over 10,000 meters per sec at 5000 K using hydrogen.

As regard the direct utilization of atomic energy, with its particle velocities of 1.2×10^7 meters per sec in uranium nuclear fission, for instance, no technical solution is in immediate prospect.

Combustion Chambers: The mixture preparation and the heating of gases take place in the rocket's combustion chamber.

In chemically propelled rockets, the mixture preparation is an exceptionally complex sequence of physical and chemical processes, such as injection, jet breakdown, droplet formation, droplet heating, droplet evaporation, turbulent mixing, mixing by diffusion, ignition, combustion, adjustment to equilibrium of the combustion gases, etc., which are at present being thoroughly studied and which can be influenced to a certain degree by constructional features. The goal of these studies is not to find methods enabling very complete combustion to be obtained, as the completeness of the combustion, owing to the high temperatures and consequently high-reaction rates, is in any case as high as the chemical equilibria in the combustion gas allow it to be, if the preparation is adequate. Otherwise, the stationary combustion would run the risk of pulsating or dying out.

The object of the studies is far more to find methods enabling the smallest possible combustion chambers to be used, which is mostly coupled with fastest possible preparation.

Time and space requirements for the combustion itself mostly give way completely to the corresponding requirements of mixture preparation.

We must begin by making a sharp distinction between single-phase liquid injection and multiphase wet-vapor injection of the fuels.

In the case of single-phase injection, the desired fast breakdown of the jet is sought with special nozzle design, e.g., perforated nozzles, slit nozzles, annular-slit nozzles, twist-injection nozzles, and so on. The theory of the twist-injection nozzles, for instance, reveals that the amount they inject and the angle of the aperture of the jet cone are determined by the cross section of the entry to the chambers, the eccentricity with which this cross section is centered, and by the cross-sectional area of the nozzle outlet.

These conditions make it possible to build a twist-injection nozzle which can be regulated within the wide limits, and for which the size of the entry cross section of the chamber can be controlled by slight modification of the injection pressure.

The theory of the single-phase and multiphase injection shows that, for the first type of jet, solely immediate impulse exchange takes place with the ambient medium mixed with it, whereas for the second type of injection, an immediate exchange of impulse and enthalpy occurs, thus enabling the wet-vapor droplets to complete their evaporation quickly, and the ignition temperature of the fuels to be exceeded, while in the first case the droplets remain cold and unvaporized for a long time.

Both types of injection also differ from each other considerably as regards droplet formation, insofar as the single-phase jet furnishes much larger droplets through dynamic pulverization, as a result of the equilibrium between surface tension and ram pressure, and as the multiphase jet forms its droplets by thermic pulverization as a result of the equilibrium between surface tension and vapor pressure. These circumstances make it seem worth while to strive for artificial thermal pulverization by addition of corresponding amounts of fuel.

The heating of the droplets is achieved quickly by convection only after thermal pulverization, because the droplets then attain a size about equal to a molecule's mean free path and, owing to the effective gas kinetic convection mechanism in mixture with burning gas, vaporize quickly, whereas the convection in dynamically pulverized droplets is only effective as long as they have not begun to boil. In the latter case, therefore, the only remaining method of heating is by radiation of combustion gas.

The theory of the three main components of the radiation of combustion gas, namely, aerosol radiation, CO_2—H_2O radiation, and chemiluminescent radiation, has been developed sufficiently to yield certain quantitative results of practical utility. The principles of the latter are based on the centi-

meter wave reflection measurements of Goerke on rocket combustion gases.

For hot hydrocarbon combustion gases, gray aerosol radiation is of no significance owing to the lack of soot precipitation as a result of the replacement of CO_2 combustion by CO combustion after adjustment of the chemical equilibrium; it is only of significance in cool combustion gases which disassociate poorly or contain metal.

The long-wave CO_2—H_2O radiation yields an intensity which amounts to fractions of the radiation of an equally hot black body, though this fraction is all the same dependent on the temperature and the thickness of the layer of the radiated masses of gas. It sometimes amounts to 15 per cent.

The short-wave chemiluminescent radiation (reaction luminescence) depends, as regards its total energy, solely on the heat-toning of the reaction taking place. Its intensity, on the other hand, increases quickly with the gas density, and can be much higher than that of an equally hot black body; accordingly, however, its range decreases with density.

In order to make the droplet-heating effective through radiation, it can become important, especially in the case of long-wave radiation of combustion gas, to suit the emission spectrum of the combustion gas to the absorption spectrum of the fuel by coloring the latter. Nevertheless, CO_2—H_2O radiation is coupled with very long heating durations, which, under suitable circumstances, can be considerably shortened by reaction luminescence.

Within the limits of the determinant time and space requirements, it is mostly the droplet vaporization and mixture which take the longest time.

The droplet vaporization can only proceed in a short time either if the fuel is injected in two phases, i.e., as wet vapor, because the convection is then effective, or if the combustion temperature and density of the gases are so high that the reaction luminescence of the combustion becomes sufficiently intense to vaporize even mechanically pulverized droplets within the first fraction of a millimeter of their trajectory.

The time and space requirements of the droplet vaporization are especially noticeable when fuels are injected in liquid form into burning gases of low density and moderate temperature, so that the heating should not be so much affected by convection or luminescence, but rather by the relatively weak but penetrant CO_2—H_2O radiation.

Only in this case is it necessary to attach particular value to fine pulverization, e.g., by utilization of twist-injection nozzles, multiphase additions to the fuel, etc., and eventually to alter the absorption spectrum of the fuel. In cases of luminescent reactive combustion gases, the fineness of the dynamic pulverization is relatively unimportant. It is the mixture itself which as a rule fixes the time and space requirements.

The most effective way of favoring it is by means of extensive primary macromixing, thus, by injecting the fuel through a great number of small and well distributed nozzles.

The mixing process is further improved by the droplet jets of the second-

ary macromixture, which cross each other; furthermore by the turbulence of the gaseous stream components, finally to be completed by diffusion (effective over only short distances).

These measures make it seem possible that the present usual ratio of combustion-chamber volume to narrowest outlet cross section, amounting to three meters in rocket and jet engines, will be cut down considerably, perhaps to a fraction of a meter.

The mixture preparation requires a finite period of time, which one must make available to the fuels in the form of their storage life in the combustion chamber. For this ratio of the combustion-chamber volume V_0 to the volume of the combustion gases passing through per sec, the following expression is obtained:

$$t = \frac{V_0}{f'}\sqrt{\frac{\left(\frac{\kappa+1}{2}\right)^{\frac{\kappa+1}{\kappa-1}}}{\kappa g R T_0}} \qquad [12]$$

whereby f' is the narrowest outlet cross section, e.g., the cross section of the neck of the nozzle, and κ, R, T, time mean values of adiabatic exponents, gas constants, and temperature of the gases in the combustion chamber. Hence, with the same fuels and the same degree of combustion, the storage life is primarily dependent on the ratio of the volume of the combustion chamber to the cross section of the neck of the nozzle, whereas the actual shape of the combustion chamber, and the combustion pressure, are largely without significance.

In the case of the liquid-fuel rockets manufactured in Germany, V_0/f' was always in the vicinity of three meters, which, under the combustion-gas conditions prevailing at the time, would correspond to a storage life t of approximately 10^{-2} sec.

As practical experience has shown, this duration, in a case of intense macromixture, is enough for mixture preparation and complete combustion.

In the case of atomic rockets, too, a corresponding time for preparation and a corresponding size of combustion chamber will be necessary.

Noteworthy is that this figure, $V_0/f' = 3$ meters, has so far proved its utility in flight tests, as the lower limit value of the volume of the combustion chambers of ramjet engines and resonance ducts, provided the macromixture was well advanced. Closer investigation of the preparation processes, however, reveals that there are important cases where the requisite combustion-chamber volume is less defined by the narrowest outlet cross section of the flame tube—roughly in the form of $V_0 = 3f'$—than by the distance d from each other of the injection nozzles—roughly in the form of $V_0 = 2.5\pi d^3 n$—where n is the number of injection nozzles, so that the combustion chambers of rockets and ramjet engines can become considerably smaller with d, and within certain limits the principle of geometrical similarity can be applied in the construction of scale models.

(*To be continued in next issue*)

U. S. ROCKET DEVELOPMENT[1]

Dr. Goddard, in whose memory this award has been established, was a personal friend for a number of years, and I have always had a great deal of admiration for his work in pioneering the development of the liquid-fuel rocket.

Dr. Goddard's work dates back to the days when the potentialities of the liquid-fuel rocket were not generally recognized. Harry F. Guggenheim was among the few who had the foresight to see the future, and he gave generous support to Dr. Goddard's ideas. Dr. Goddard's work was well in advance of the Germans, even though some of it had to be done under adverse conditions. With a modest staff of about six men and his limited facilities, by 1940 he had succeeded in designing and constructing what was virtually a prototype of the German V-2 rocket on a small scale, a project upon which the Germans had at one time employed as many as 20,000 men and expended a sum of 300 million marks!

The major part of effort at present is directed toward fundamental research underlying the basic principles of rockets and guided missiles. This approach is aimed principally at ultimate perfection and constitutes an excellent long-range program.

However, what is urgently needed now is the extension of this program to put much greater emphasis on the reduction to practice. We need large-scale testing of the various rocket devices in order to work out production methods, to train ground crews, and to gain operational experience with these missiles. It is well known that an experimental engine cannot be converted to a production unit in much less than a year or even longer.

We have at present at least six rocket engines and missiles which operate reasonably well, as, for instance, the Gorgon, the Lark, the Viking or Neptune, the Wac Corporal, the Aerobee, and some others. These units could be the nucleus for the enlarged program.

It is doubtful that we are completely familiar with the progress of rocket development in other countries. A great deal of the German technical information and a number of scientists have passed into other hands, and it may well be that they have taken a different line of approach in solving their problems, in which case they may have a considerable lead on us in the reduction to practice of flight articles.

It behooves us, therefore, to look to the present as well as the future and to parallel our fundamental research program with a much greater emphasis upon getting practical applications and equipment for field use, even though it may not be the ultimate design.

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[1] Address by John Shesta, Mem. ARS, Reacton Motors, Inc., Dover, N. J., on occasion of his acceptance of the Goddard Memorial Lecture Medal conferred for the first time at the 1948 Annual Dinner of the American Rocket Society, Hotel Statler, New York, N. Y., Dec. 2, 1948. Slightly condensed.

AMERICAN ROCKET SOCIETY ELECTS FIRST FELLOW MEMBER

IT HAS long been the desire of the American Rocket Society to establish a Fellow classification within its membership ranks, and the decision has been made to elect worthy individuals to this classification from time to time. The Board of Directors of the American Rocket Society are honored to announce the election of the first Fellow of the Society, Dr. Theodor von Kármán, professor of aeronautics and director of the Guggenheim Aeronautical Laboratory of the California Institute of Technology.

THEODOR VON KÁRMÁN, FIRST FELLOW OF THE AMERICAN ROCKET SOCIETY

Dr. von Kármán was born in Budapest, Hungary, in 1881. He first visited the United States in 1926 and returned for permanent residency in 1930. Since that time, he has been professor of aeronautics and director of the Guggenheim Aeronautical Laboratory of the California Institute of Technology.

Before coming to the United States, Dr. von Kármán had a distinguished career in engineering. He graduated as a mechanical engineer from the Royal Technical University of Budapest and received his Ph.D. from the University of Goettingen, Germany. He has been awarded the honorary degree of doctor of engineering by the Technical University of Berlin, and Princeton University; an honorary degree of doctor of science from the University of Brussels; and an LL.D. from the University of California.

He is a member of the Royal Society of London, the French Academy of Sciences, Accademia dei Lincei in Rome, the Royal Academy of Sciences of Torino, the National Academy of Science, Honorary Fellow of both the Royal Aeronautical Society and the Institute of Aeronautical Sciences, Honorary Member of the Italian Association of Automotive Engineers and the French Society of Civil Engineers. He recently received the Franklin Medal and the John Fritz Medal.

He is consultant to the Northrop Aircraft Company, the General Electric Company, and the Aerojet Engineering Corporation. He is chairman of

the Scientific Advisory Board to the Chief of Staff, United States Air Force.

In the rocket field, Dr. von Kármán, with a small group of younger associates, began rocket research in 1937 at the California Institute of Technology. The GALCIT Project, which later became the Jet Propulsion Laboratory of the California Institute of Technology, grew from this rather modest beginning, as did the Aerojet Engineering Corporation of Pasadena and Azusa, Calif. The first long-range rocket project in the United States was sponsored by the Army Ordnance Department in 1943, as the result of a joint memorandum to the Chief of Ordnance from Dr. von Kármán, Frank Malina, and H. S. Tsien. In 1938, Dr. von Kármán initiated the planning for the first supersonic wind tunnel in the United States.

Dr. von Kármán, in accepting the grade of Fellow expressed his appreciation of the honor, and stated that he regretted "that Dr. Goddard was not among us to occupy this number one post, which would certainly be due him."

JET PROPULSION CENTERS AND FELLOWSHIPS ANNOUNCED

AN EVENT of far-reaching significance in the field of jet propulsion was announced jointly on Dec. 16, 1948, by Princeton University, California Institute of Technology, and The Daniel and Florence Guggenheim Foundation. Two Guggenheim Jet Propulsion Centers have been established, one at each of the two institutions, to provide for postgraduate education and research in the field, with particular emphasis on peacetime uses.

The Daniel and Florence Guggenheim Foundation has appropriated $500,000 to support the Centers for a period of seven years. These funds are to be used to pay salaries of professors, stipends of graduate students, and similar expenses. Necessary buildings and equipment are to be provided by the universities.

To Honor Rocket Pioneer

The principal post in each of the Jet Propulsion Centers will be a Robert H. Goddard Professorship, named in honor of the American rocket pioneer. Each Goddard Professorship will be associated with three postgraduate fellowships, to be known as the Guggenheim Jet Propulsion Fellowships, and will carry a stipend of up to $2000 each.

Located at eastern and western institutions which are already established centers of research and training in the field and in allied fields, the Jet Propulsion Centers have a threefold purpose: (1) To train leaders of the

future in rocket and jet-propulsion technology; (2) to promote research and advanced thinking on rocket and jet-propulsion problems; and (3) to serve as centers of leadership in the development of peacetime commercial and scientific uses of rockets and jet propulsion.

The Robert H. Goddard Professorships are to be filled by outstanding men of international reputation in this engineering field. The Robert H. Goddard Professor at California Institute of Technology will be H. S. Tsien, now professor of Aerodynamics at Massachusetts Institute of Technology. The Robert H. Goddard Professor at Princeton University is to be announced shortly.

The fellowships are open to qualified young men who are suitably prepared for postgraduate study in this field. Applicants will normally be expected to have a bachelor's degree from an accredited institution, in some branch of science or engineering, and to show promise of exceptional ability to pursue effectively graduate study and research in jet propulsion.

Candidates must be residents of the United States, must have outstanding technical ability, deep interest in the development of rockets and jet propulsion, an intention to follow this field as a career, and qualities of leadership.

Postgraduate Fellowships

The fellowships are for postgraduate study only. They are normally awarded for two years each, but are renewed at the end of the first year only when the work performed by the fellow is satisfactory to the university. In unusual cases, at the discretion of the university, a longer period may be allowed. While no age limit has been set, candidates should preferably be under thirty.

Establishment of the Centers is in line with the Foundation's long interest and association with rocket and jet propulsion development. This association dates from the 1930's when the Foundation financed Dr. Goddard's brilliant work which laid down the basis for much of the rocket progress made during the war and since.

After the first World War, the Daniel Guggenheim Fund for the Promotion of Aeronautics contributed materially to commercial aviation development in the United States when such development was in its infancy. In establishing the new Jet Propulsion Centers, the Foundation is endeavoring to contribute in a somewhat similar way to the development of peacetime applications of jet propulsion.

These two endeavors cannot be regarded as strictly parallel, however. The peacetime future of jet propulsion development seems at once less clearly definable yet broader than was the future of aeronautics after World War I.

The growth of both fields has the same twofold ends—the evolution

of useful devices, and the advancement of a portion of the frontier of scientific knowledge. It is in the first end that the future of jet propulsion seems the less clearly definable, especially if considered only as a segment of the field of aeronautics. While the family of jet-propulsion power plants can together fulfill almost any demand for speed in aircraft, their use imposes successively greater problems in boost, range, deceleration, and operating economy. Typically, acceleration to the regimes in which the jet engines are effective requires energies whose magnitudes may approach or exceed any steady-state requirements. Economical solutions to such problems may or may not be forthcoming. Paradoxically, one of the primary peacetime uses for the rocket engine may be as an assist-take-off boost on aircraft powered by conventional engines.

Meteorological Applications

Some of the jet family have potentially useful applications in realms other than commercial aviation. One of the greatest hopes is for the meteorological sounding rocket or ramjet. The uncertainty here lies not with the devices but with the development of meteorological science and technique to a point where accurate long-range weather predictions might be made on the basis of high altitude observations. Such weather predictions will most certainly be of immense economic importance. Two little-known offshoots of jet propulsion which may develop into devices of major importance are the hydropulse and the hydroduct. The potential broadness of their use must rest with the findings of their investigators.

It is thus that the future of jet-propulsion devices is difficult to define. It is in the second end—the advancement of the frontier of scientific knowledge—that the future of the jet propulsion sciences seems broad. The advent of the turbojet, ramjet, and rocket has forced a revolution in engineering concepts of the conversion of energy into propulsive force. The new perspective reveals new frontiers and reillumines old ground. Second, jet propulsion demands information from what have been the nether regions of the sciences which compose jet propulsion; such future information will be applied in many fields other than jet propulsion. Third, because the field does embrace many sciences, its expansion can be expected to bring to light basic interrelationships and facts of broad usefulness in such branches as fluid mechanics, heat transfer, thermochemistry, and metallurgy.

Thus the primary accomplishments to be expected from the personnel of the Guggenheim Jet Propulsion Centers appear to lie less in promoting useful devices than in supplying scientific perspective and stimulation to the field, in obtaining key technical information, and in seeking and making known fundamental laws which may involve the several sciences. Their task will require judgment and inspiration.

Third Annual Convention Held in New York

THE Third Annual Convention of the American Rocket Society was held in conjunction with The American Society of Mechanical Engineers at the Hotel Statler, New York, N. Y., Nov. 28 to Dec. 4, 1948. The Convention was an outstanding success, not only because the registration for the ARS almost doubled that of the last meeting in New York, but also because the growth of the ARS was evidenced in the Convention activities themselves.

Technical Sessions

The ARS presented two technical sessions, a luncheon, and, in co-operation with the Aviation Division of the ASME, a dinner. Two papers were presented on Thursday morning, Dec. 2. Charles H. Harry, pilotless aircraft section, Glenn L. Martin Company, Baltimore, Md., spoke on "A Study of the Parameters Affecting Over-All Rocket Performance"; and B. N. Abramson, Bell Aircraft Corporation, Buffalo, N. Y., presented a paper on "A Survey of Injector Designs for Use in Liquid Propellant Rocket Motors." On Thursday afternoon, C. J. Turansky and R. D. Rinehart, rocket group engineers, Bell Aircraft Corporation, Buffalo, N. Y., presented a paper on "Creation of High-Pressure Gas Source for Rocket-Motor Propellant-Supply Systems," and L. G. Bonner, technical director, Allegheny Ballistics Laboratory, Cumberland, Md., spoke on "Metal Parts for Solid-Propellant Rockets." Both sessions were well attended. The luncheon speaker was K. F. Mundt, chief engineer, Aerojet Engineering Corporation, Azusa, Calif., who gave an interesting illustrated lecture on "The Muroc Facility."

Awards Dinner

In co-operation with the Aviation Division of the ASME, the American Rocket Society held its annual Awards Dinner. This dinner marked the beginning of an annual presentation of three American Rocket Society awards. The Goddard Memorial Award, a gold medal, was made this year to John Shesta, Reaction Motors, Inc., for his outstanding work at Reaction Motors, Inc., and his long-standing interest in the American Rocket Society. The award of a silver medal for the C. N. Hickman award was made to Frank Malina of UNESCO for his original work in assisting in the setting up of GALCIT and in the founding of Aerojet Engineering Corporation, and for his outstanding work on the WAC Corporal.

The third award, a bronze medal, for the American Rocket Society Junior Award was presented to four students at College of the City of New York for a paper on their rocket project called "Rocket Thrust Cylinder Design and Experimentation." These students were Noel Rothmayer, Arthur Sherman, Abe Bernstein, and Frederick Linzer. The awards were made by G. Edward Pendray, toastmaster, who also presented the speaker of the evening, Hugh L. Dryden. Dr. Dryden, director of aeronautical research, National Advisory Committee for Aeronautics, Washington, D. C., spoke on the subject, "Rockets as Research Tools in Aeronautics."

Election of Officers

The casting of the mail ballots for election of officers for the national organization of the American Rocket Society resulted in the following officers being elected for the year ending January, 1950: President, William L. Gore; vice-president, Leonard Axelrod; directors for a term of three years, R. Tom Sawyer, John J. Murphy, and Lieut. Comdr. F. C. Durant, 3d.

New York Section

THE casting of the mail ballots of the New York Section for the year ending January, 1950, resulted in the election

of the following members to the Board of Directors:

For a three-year term, Morton Gerla, James Wheeler, and James Nutt, Jr.; two-year term, Henry Georgious, Thomas S. Gardner, and Lawrence Heath; one-year term, Frank A. Coss, Edward Schwartz, and W. E. Kuhn.

At the first regular meeting of the new Board of Directors held in January, 1949, in accordance with the By-Laws approved by the membership, the following officers were appointed: President, Morton Gerla; vice-president, James Wheeler; treasurer, Frank A. Coss; and secretary, James Nutt, Jr.

The New York Section had two very successful meetings this year. The January meeting was addressed by Maurice J. Zucrow, professor of gas turbine and jet propulsion, Purdue University. Dr. Zucrow gave an interesting illustrated lecture, and in spite of bad weather the meeting was well attended. The second meeting this year was a joint meeting held with the Institute of Aeronautical Sciences at which R. E. Gibson, director, Applied Physics Laboratory, Silver Spring, Md., was the main speaker. A board of experts consisting of A. Kossiakoff, associate director of the Applied Physics Laboratory, W. H. Goss, director of the Bumblebee Project, F. T. McClure, APL rocket expert, Rear Admiral Frederick I. Entwhistle, guided missile chief, Bureau of Ordnance, and Henry K. Porter, supervisor, Bumblebee Laboratory Unit, answered all questions presented by the audience after the illustrated talk was completed.

The attendance at this meeting was over 600 persons and both the IAS and the ARS felt the evening had been particularly successful.

Southern California Section

THE American Rocket Society is happy to announce the establishment of a Southern California Section in Los Angeles, Calif. The considerable activity in this field in the Southern California area led a group of engineers to form a technical organization for the advancement of all engineering phases of jet propulsion. It was the decision of this group to establish the Southern California Section of the ARS. The national organization feels certain that the activities and contributions of this Section will do much to add to the already strong position held by the ARS in the field of jet propulsion. The first meeting of the Southern California Section was held on Feb. 17, 1949, at the Park View Playground Building, Los Angeles, and the speaker, Colonel B. S. Mesick, United States Ordnance Department, discussed military research contracts. The meeting was an organizational one and nominations for officers were made. A full report on these nominations will be made in the next issue of the JOURNAL. The second meeting of the Southern California Section will be held in conjunction with the ASME on April 28, 1949, at the Southern California Gas Company Auditorium, 810 South Flower Street, Los Angeles.

At that time Walter Reidel will speak on "The Status and Direction of German Rocket Development at the End of World War II."

Washington, D. C., Section

THE formation of a Washington, D. C., Section of the American Rocket Society has met with much enthusiasm in that district, and it is a pleasure to announce that the first meeting of the Section was held on Monday, March 7, 1949, in the Washington Gas Light Auditorium. Lieut. Comdr. Robert C. Traux, Bureau of Ordnance, U.S.N., presented a talk on "The Pioneer Rocket Project of the U. S. Navy." The meeting, held in co-operation with the Engineers Club of Washington, was well attended. The national organization feels certain that the Washington, D. C., Section will contribute a great deal to the improvement of the Society.